NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
CONTRACT NO. NAS 7-100

SEMINAR PROCEEDINGS

MECHANICAL DESIGN OF SPACECRAFT

MARCH 28–MAY 23, 1962

N67 11281

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

August 1, 1962

FOREWORD

James L. Adams

In the spring of 1962, engineers from the Engineering Mechanics Division of the Jet Propulsion Laboratory gave a series of lectures on spacecraft design at the Engineering Design seminars conducted at the California Institute of Technology. Several of these lectures were subsequently given at Stanford University as part of the Space Technology seminar series sponsored by the Department of Aeronautics and Astronautics. Presented here are notes taken from these lectures.

The lectures were conceived with the intent of providing the audience with a glimpse of the activities of a few mechanical engineers who are involved in designing, building, and testing spacecraft. Engineering courses generally consist of heavily idealized problems in order to allow the more efficient teaching of mathematical technique. Students, therefore, receive a somewhat limited exposure to actual engineering problems, which are typified by more unknowns than equations. For this reason it was considered valuable to demonstrate some of the problems faced by spacecraft designers, the processes used to arrive at solutions, and the interactions between the engineer and the remainder of the organization in which he is constrained to operate.

These lecture notes are not so much a compilation of sophisticated techniques of analysis as they are a collection of examples of spacecraft hardware and associated problems. They will be of interest not so much to the experienced spacecraft designer as to those who wonder what part the mechanical engineer plays in an effort such as the exploration of space.

CONTENTS

N67 11282

Science and Engineering in Space

W. J. SCHIMANDLE

Chief, Spacecraft Development Section

Those of us who are associated with the spacecraft business are extremely interested in engineering education. This is a pointed interest, since in the long pull, we depend on schools to provide people to help us solve our rather generous supply of problems. The exploration of space is resulting in one of man's greatest technical challenges. To meet this challenge we need engineers who can handle engineering tools, make decisions based on limited data, and most important of all, *think*.

The greatest difficulties in the space program arise not because of the failure of scientists to provide abstract theory, but because of failures in physical hardware. Space vehicles are presently extremely susceptible to freezing LOX valves, random electronic failures, and other miscellaneous engineering errors or omissions. These problems can generally be traced to poor design procedure or execution rather than to lack of knowledge about the basic character of the physical universe.

This is not to say that engineering is more important than science. The scientist must discover truth, and the engineer must use it. Both scientists and engineers are required to solve the immense problems of collecting resources and operating on them in order to accomplish a task as large as the exploration of space. We are confronted with a situation that requires a balance of application of scientists and engineers, and we need the assistance of the universities in preparing people for both functions in our constantly growing and evolving programs. We must be careful that students do not imagine it to be a better "deal" to be scientists than engineers, or vice versa, for both are needed.

In the past few decades engineering education has changed a great deal. Analytical treatments are gaining favor over empirical treatments as a basis for engineering judgments. Technological achievement has succeeded in attaining a position of unprecedented importance in our culture. The effects have been both good and bad from an engineering education standpoint.

The great amount of publicity in the press has tended to emancipate the technical person from the "egghead" status. Considerable funds have been made available for scientific research through the efforts of the Government and various other organizations. The evolution to more dependence on the physics and less on the handbook has definitely fostered a greater understanding of "what is going on" among engineers.

However, on the negative side, engineering curricula now tend to emphasize science to the exclusion of basic engineering tools such as drawing. Funds poured into science areas, coupled with the unfortunate tendency of the press to label everyone a "scientist," result in a "glamour unbalance," which is enticing an ever-increasing number of students into pure science with a subsequent drain on engineering enrollments. Even more seriously, the over-emphasis on engineering science is producing people who are as hypnotized by analysis as were the handbook engineers hypnotized by gears, clutches, and bolts. The entire critical area of synthesis is being neglected in favor of an overwhelming emphasis on analysis.

It occurred to us that by taking our problems, explanations of techniques, and achievements directly to you, both your plans and our programs could be helped. The lectures and demonstrations that have been scheduled will give you an idea of what we encounter in spacecraft programs and some concept of the type of engineering problems posed by interplanetary travel. It should become clear to you that in an engineering problem, a process and method of thought exists, and this thought is affected by the position of the thinker. Since those of us who are giving this series of lectures are from the Engineering Mechanics Division of the Jet Propulsion Laboratory, we will be talking only as men involved in spacecraft mechanical engineering, even though what we say will be more or less generally applicable.

Today's lecture is entitled "Science and Engineering in Space." It is intended to be a general treatment covering

various aspects of the spacecraft engineer's task, and some background and information related to this task. Space environments will be treated lightly.

With the successful launching of *Sputnik* by the Russians, space exploration methods changed suddenly from the theoretical, based on observations from points on or near the Earth's surface, to experimental, based on measurements taken in space. Now we are in the process of developing rockets that will take us away from the Earth, into regions immediately surrounding the Earth, to the Moon, and to regions in the vicinity of Mars and Venus. Eventually, we shall go out of the plane of the ecliptic, towards Mercury and past Mars through the asteroid belts to Jupiter and other planets. The amounts of energy required for these later trips are very large. It will be some years before development of new propulsion systems will allow this expansion of our exploration program.

The engineer's task is to design a spacecraft for a specific mission. Figure 1 is a simplified plan or procedure for developing such a design.

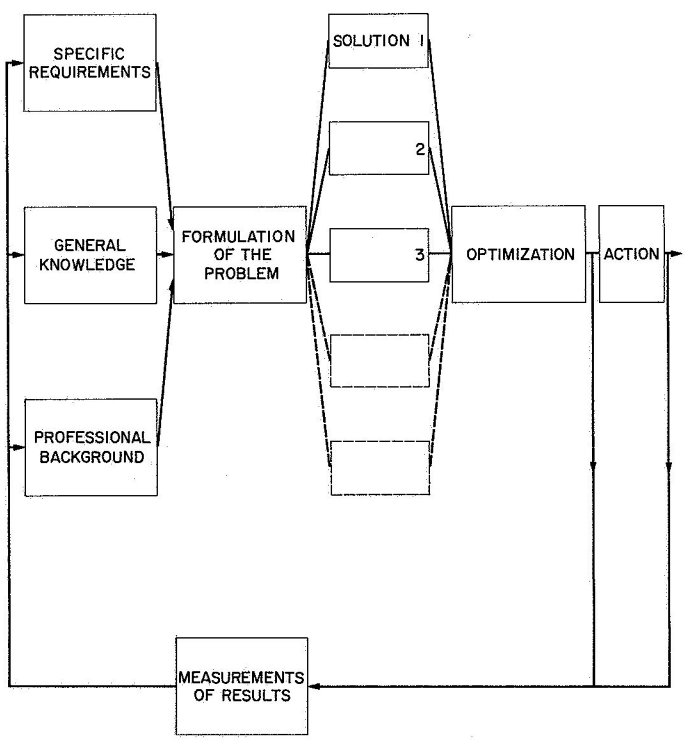


Fig. 1. Block diagram of the engineering task

The first block indicates that a specific requirement must exist. In this case it was decided that Venus would be closely investigated during 1962. To accomplish such a task, two things are initially done. First, we draw on our professional background, judgment, and experience. Second, we investigate general knowledge on the subject. At JPL we have knowledge concerning trajectories, propulsion, guidance and control, communications, and other relevant engineering and scientific disciplines. We also have a considerable background that assists us in the application of that knowledge.

The most difficult step in the process is the formulation of the problem. It is here that we are faced with the fact that in these new fields we do not have adequate knowledge, and that few precedents have been established on which we can rely for solutions. We have no proven experiences to offer guides to our progress. We are forced by the nature of our work to make assumptions, some of them very broad, which commit vast resources in men and money.

After formulating the problem, attempts are made to find solutions. In engineering, we know that there is no single solution to any problem. There are always multiple solutions based on different emphasis of the importance of various parameters. Unlike your texts, there is no concise statement and no answer book. There is also the ever-present problem of personalities and resulting conflicting value judgments. It is also not unusual to have ideal technical solutions displaced by other non-technical requirements.

After many solutions are developed, we enter into that phase of the design process called optimization, or "horse-trading." During this period the pieces are hammered together into a whole. Compromises are made as necessary so that the design becomes harmonious. The engineer then carries the proposed solution into physical being, the hardware is tested, and the results fed back into the beginning of the process. The engineer carries his design around this loop until, finally, an acceptable solution is achieved. It is this process of synthesis, analysis, and evaluation that characterizes all engineering problems, whether they be expanded to cover the universe or limited to the production of a pin. The process is unchanged except by degree.

Figures 2 and 3 give you some impression of the difficulties of space exploration.[1] The solar system is indeed large and the environments harsh. It is not accidental that it has taken man so long to go into space. His sophistication in the use of technical and managerial tools had

[1]Fig. 3 is reproduced here by courtesy of C. S. Hammond & Co., Inc., N. Y.

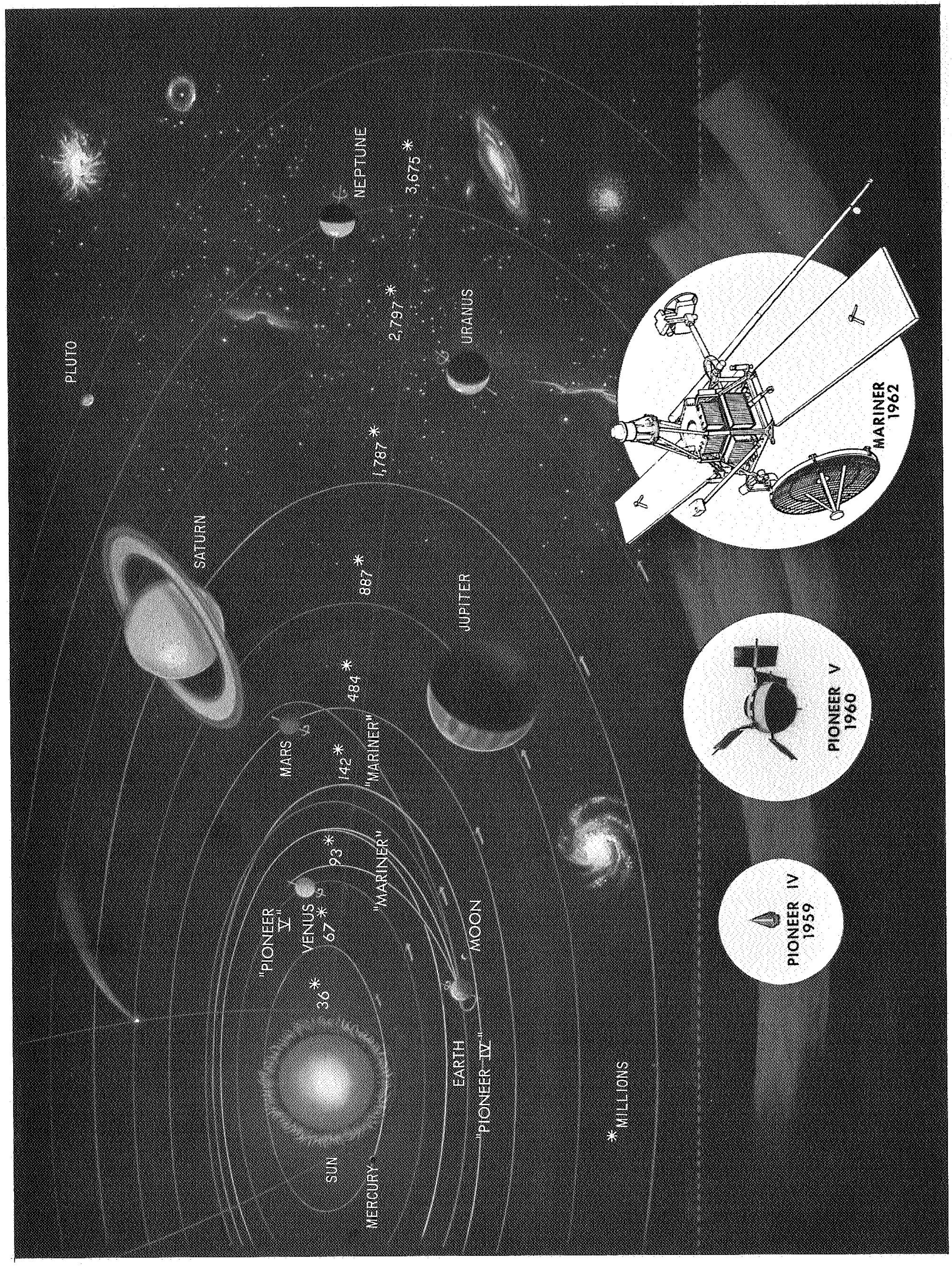


Fig. 2. Planetary exploration

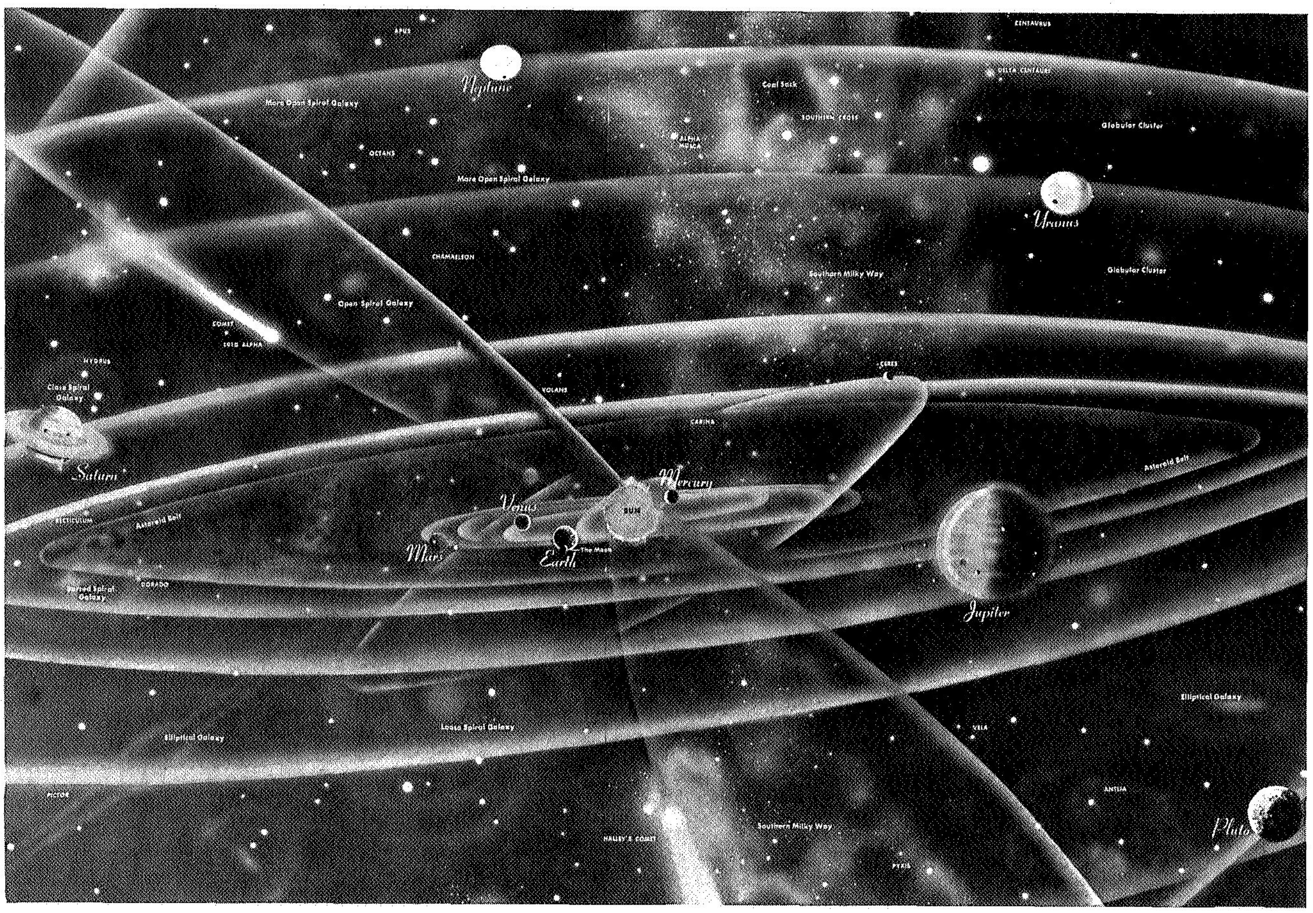


Fig. 3. The solar system

to grow to a high level to permit the tackling of such an enterprise.

The size of our solar system, even though infinitesimal by cosmological standards, strains our understanding and technical capabilities to the limit. As shown, the planets revolve in a counterclockwise direction around the Sun.

The Earth's orbital plane is called "the plane of the ecliptic" and the Earth revolves at an average distance of 93 million miles. Mercury is at 36 million, Venus at 67 million, Mars at 142 million, and Pluto at 3,675 miillion.

Each planet revolves in a plane which is inclined to the ecliptic. This is shown in Fig. 4 for one specific period in time and with all orbits projected on a plane.[2] This, of course, is not the real case because the orbital planes do not intersect in a line and the orbital inclinations are subject to small variations because of the perturbations of other planets. It is interesting to note that the inclinations of all orbits are small except that of Pluto.

A typical interplanetary and lunar launch trajectory is shown in Fig. 5. The probe is launched into an orbit around the Earth. The engines are turned off when orbital velocity is achieved and the vehicle is allowed to coast or "park" until it reaches an optimum energy position to add that increment of velocity which will permit escape. At this point, the engines are reignited and the spacecraft injected on the proper flight path. Such a technique permits maximum utilization of payload capability over an acceptable firing period.

Figure 6 illustrates typical heliocentric transfer trajectories for 1962. It shows a trajectory from Earth to Venus and one from Earth to Mars. Note the positions of the Earth and Venus at launch time, and then after the 108-day transit time. A vehicle following the Venus trajectory is slowed down from the Earth's heliocentric velocity by adding velocity against the Earth's direction

[2]Fig. 4 is reproduced from *Space Technology,* Howard S. Siefert, Editor, 1959, by courtesy of the publishers, John Wiley & Sons, Inc., N. Y.

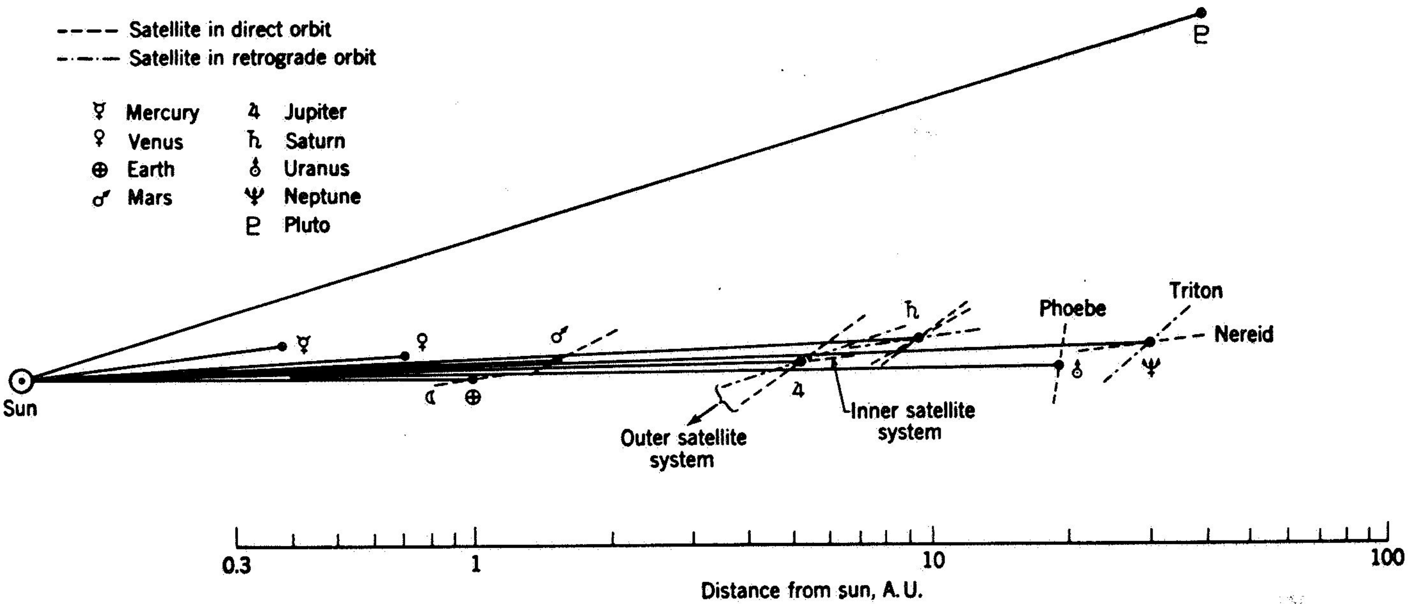


Fig. 4. Orbital inclinations of planets and their satellites in the solar system

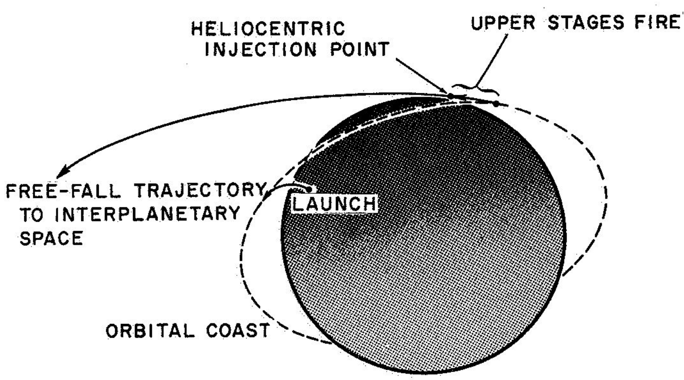


Fig. 5. Typical trajectory for chemically propelled spacecraft

of travel. In both cases these probes act like new planetary bodies assuming heliocentric orbits around the Sun, but doing so in such a manner that they will intercept the path of Mars and Venus when the planets reach their respective intercept positions. A factor to be considered in looking at this picture is that it is two-dimensional. Venus and Mars are actually in slightly different orbital planes so that the probe has to ride out of the ecliptic in order to meet the planets on their trajectories.

Figure 7 is a chart indicating relative planet distances and periods. The Earth's orbit lies on the zero line. Distances in millions of kilometers from the Earth are plotted vertically. Venus approaches the Earth within about 50 million kilometers in June of 1962, and Mars reaches its nearest proximity somewhat later in the same year. The huge periodic variation in distances represents a serious problem in space exploration. Schedules must be met. The planetary probes must be fired promptly, since we do not presently possess a surplus propulsion capability.

Figure 8 shows a typical Venus encounter. The spacecraft passes on the sunny side of Venus. One of the biggest problems, discovered early, was that we could not trust our two-dimensional intuition about planetary inter-

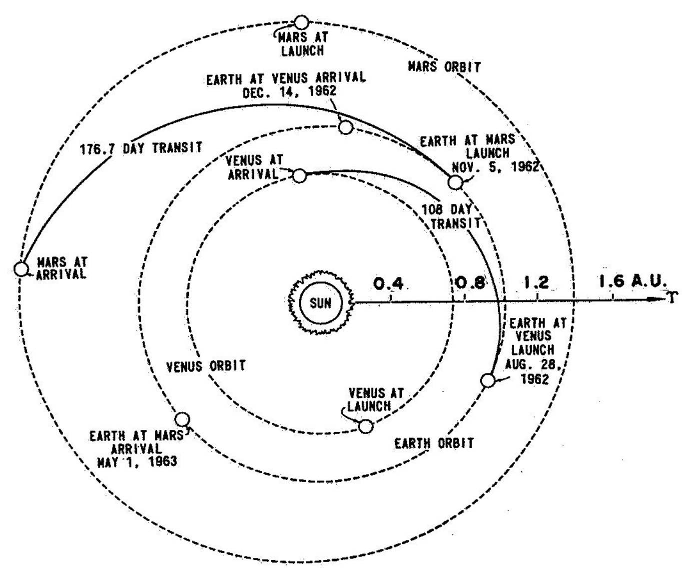


Fig. 6. 1962 heliocentric transfer

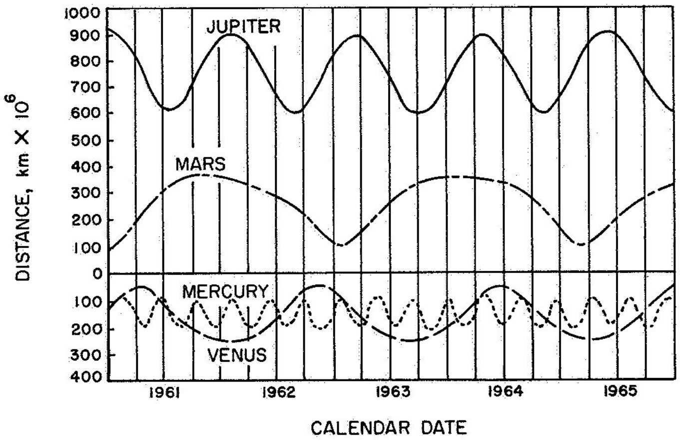


Fig. 7. Relative planet distances and periods

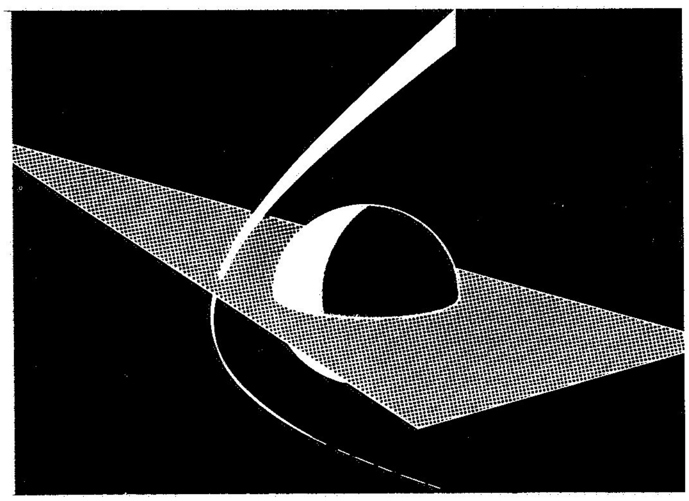


Fig. 8. *Mariner* prototype near-Venus trajectory

cept geometry. In this Figure, the Sun is at the left. The probe follows the transfer trajectory which comes in high over Venus, cuts down to the orbital plane, and disappears in a direction downward and towards the Sun. Other transfer and intercept geometries must be carefully evaluated by the "look angle" method.[3] These look angles cause severe constraints on the design of the spacecraft.

Up to this point only intercept trajectories have been considered. If it is desired to place a spacecraft in orbit around a planet, the trajectory shown in Fig. 9 must be flown. To accomplish an orbit at the planet, it is necessary to sufficiently decrease the kinetic energy of the probe so that it can be captured. This is accomplished by delivering impulse from a retrorocket. If entry into the atmosphere or descent to the surface is desired, a trade-off can be made between aerodynamic-drag devices and the retropropulsion system. To accomplish the missions described, many system elements are required. A few will be described to give you some idea of the complexity of the total job.

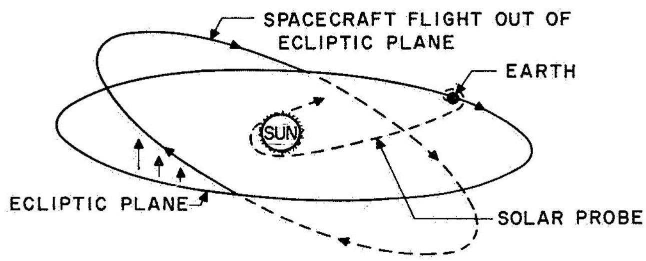


Fig. 9. Typical mission trajectories

In order to reach orbital and escape velocities, it is necessary to construct a device that is capable of converting chemical or other forms of stored energy into kinetic energy. The simple rocket is such a device, and the multistage *Atlas-Centaur* rocket shown in Fig. 10 represents a sophisticated development of this device.

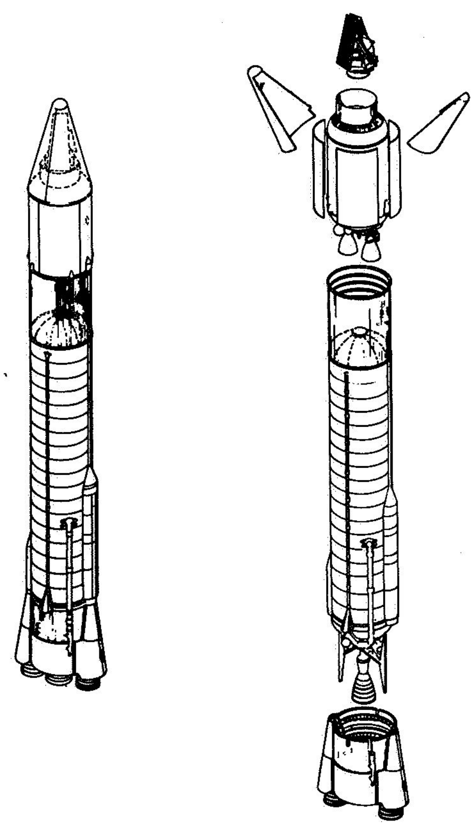


Fig. 10. *Mariner Centaur* configuration

[3]Wilson, J. N. and Coyle, G. G., *A Design Solution to the Spacecraft "Look Angle" Problem*, Technical Report No. 32-311, Jet Propulsion Laboratory, Pasadena (in preparation).

The rocket consists of a main or booster stage, a sustainer stage, and the *Centaur* stage. The rocket lifts off with the booster and sustainer-stage engines burning. At a precisely calculated point along the trajectory the booster engines, associated structure, and auxiliary equipment are jettisoned. This is done to reduce the net amount of mass—structural mass fraction—which must be accelerated during the remaining burning time of the sustainer engine. The sustainer stage continues to burn to a predetermined suborbital velocity, at which point it cuts off and separates from the *Centaur*. The *Centaur* is then ignited, and the combination of rocket and spacecraft are propelled to orbital velocity. Utilizing the parking orbit described earlier, the *Centaur* stage is again ignited and burned, which injects the spacecraft into a transfer trajectory.

The design of such a rocket is complex, and is well covered in the literature. It is of interest to note by comparison that such a rocket at take-off weighs in the neighborhood of 300,000 lb, while the payload it delivers at the injection point is in the neighborhood of 1500 lb, for an interplanetary trajectory. This vehicle gross weight to payload weight ratio of 200 to 1 gives some indication of the difficulty of the task.

In addition to the rocket, it is necessary to accomplish its launching and monitor its performance during the injection phase of the flight. This is done by means of an extensive launch-complex and down-range tracking system.

After the spacecraft is injected and during its coast phase of flight, communications must be maintained to monitor performance of its subsystems and send it instructions and information. To accomplish this task, a communications and tracking system that circles the globe has been established. This system must be manned and tied to a master control center at JPL.

The final major element of this system is the spacecraft. The spacecraft is that element which flies after injection along the trajectory to the target planet. It carries many subsystems that permit it to function, collect data, and transmit these data back to Earth.

Figure 11 is a photograph of a model of a *Mariner 1* Venus flyby spacecraft, which will be launched during the summer of 1962. It weighs about 450 lb, and carries instruments that measure the intensity of interplanetary radiation, magnetic fields in the vicinity of Venus, and Venusian temperatures. It is a simplified version of an earlier prototype, which weighed 1100 lb. The design process on the prototype took several months. It carried a long magnetometer boom, larger solar panels, a planetary scanning boom, and 19 scientific experiments.

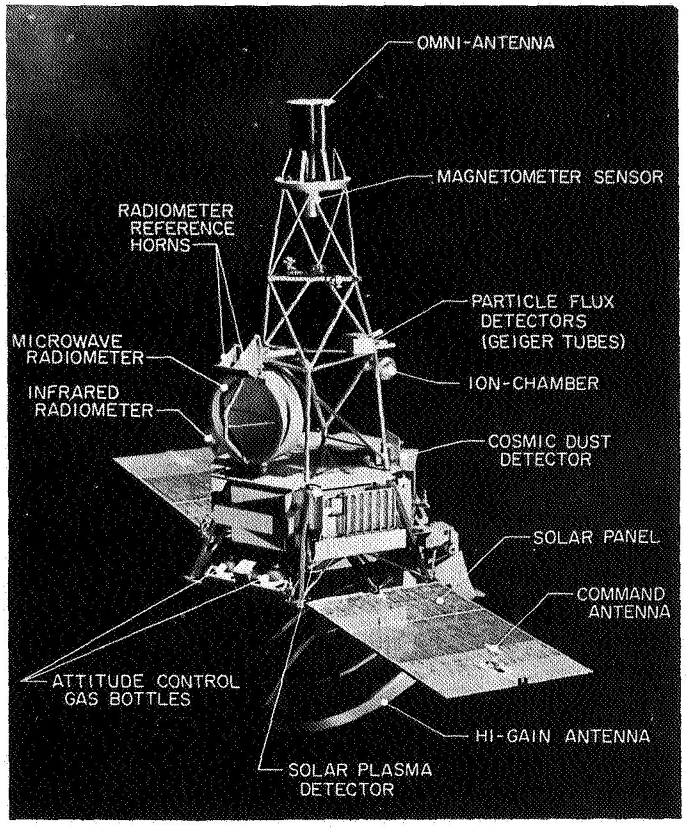


Fig. 11. *Mariner 1* spacecraft

In its journey from the launch pad to some predestined rendezvous in space, the spacecraft sees many varied environments. These are listed in summary fashion in Fig. 12 and will not be developed here. It is interesting that three of these "environments" represent areas of new

1. ENGINEERING MECHANICS	2. SPACE
a. BOOST	a. VACUUM
b. ZERO G	b. LONG LIFE
c. ENTRY	c. COSMIC DUST
d. LANDING	d. RADIATION
	e. DISTANCE
	f. THERMAL ENERGY
	g. PLANETARY ATMOSPHERE

Fig. 12. Spacecraft environments

technologies. These three are: long life, vacuum, and thermal environment. Each of these topics will be discussed in considerable detail in following lectures.

To summarize briefly, I hope you have gained some appreciation of the magnitude of the task facing engineers associated with the space exploration program. There is much work to be done. Most of this work will follow the design process described, and the problems encountered will have to be solved as they arise. There is no easy way to the solution of engineering problems. They have to be faced with a bag of tools, a creative mind, and a determined will that will not accept a second-rate approach. The spacecraft engineer cannot supply a spare part to a device that fails in space. He must be right the first time, and on a predetermined schedule.

We hope this series will give you some insight into these problems, and permit you to judge for yourselves if the tools you are being given are adequate to the job.

Spacecraft Design

JAMES N. WILSON

Supervisor, Spacecraft Development Group

Spacecraft design is different from other types of design in detail only. I will attempt to describe the design process as applied to *Mariner*, from conception through last-minute panics. I will describe a spacecraft, show how the design was reached, and present some of the problems that emerge, and their solutions.

The first contact we, as the public, have with most new products is when we read or hear about their functioning successfully, or sometimes, failing. A description of the spacecraft immediately after launch might point out that the *Mariner 1* spacecraft was launched with an *Atlas-Agena* towards Venus. It will take about 3 months for the spacecraft to reach Venus, where it will make scientific measurements of temperatures in the atmosphere and on the surface. The spacecraft will fly by the planet and continue in orbit around the Sun.

If you look at the spacecraft in a technical fashion, you will discover that the spacecraft is to be launched all folded up like a flower, having been fitted inside the nose-shroud on the booster (Fig. 1). After separating from the booster and tumbling end over end through space, it will then unfold solar-power panels, which will be used for the collection of solar energy that will be converted into electric power. Then, by means of tiny, very low-force gas jets, and some logic from small Sun sensors located on the spacecraft, it will start searching around until it finds the Sun. The next operation will be to fold out a parabolic high-gain antenna with an Earth sensor attached to it. The spacecraft will then rotate about the Sun axis until this sensor finds the Earth, whereupon it will stop. The spacecraft will then be attitude-stabilized in space. Figure 2 shows the *Mariner 1* spacecraft in the cruise configuration.

One exception to this is that after about eight days out a vernier velocity correction will be made to improve the trajectory accuracy. At that time, the antenna is pushed out of the way of the rocket-motor exhaust. The entire spacecraft is programmed through two single-axis turns

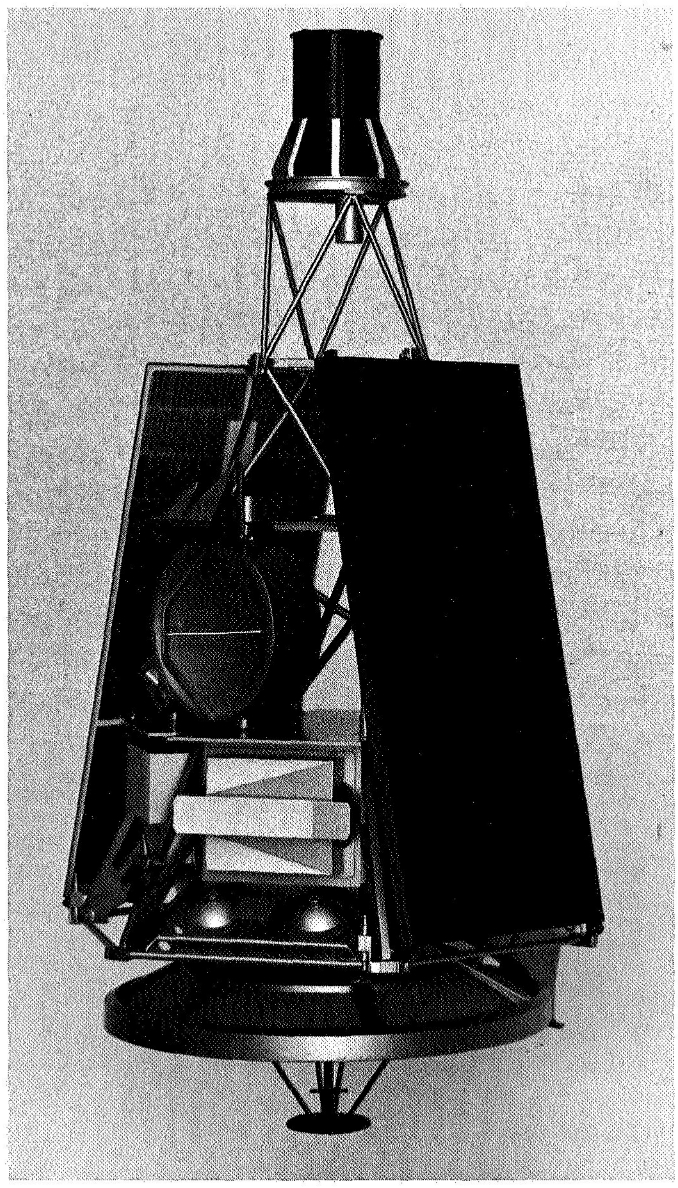


Fig. 1. *Mariner 1* spacecraft in the launch configuration

transmitted to it from the ground, and the rocket motor is fired to get the proper velocity increment. Following

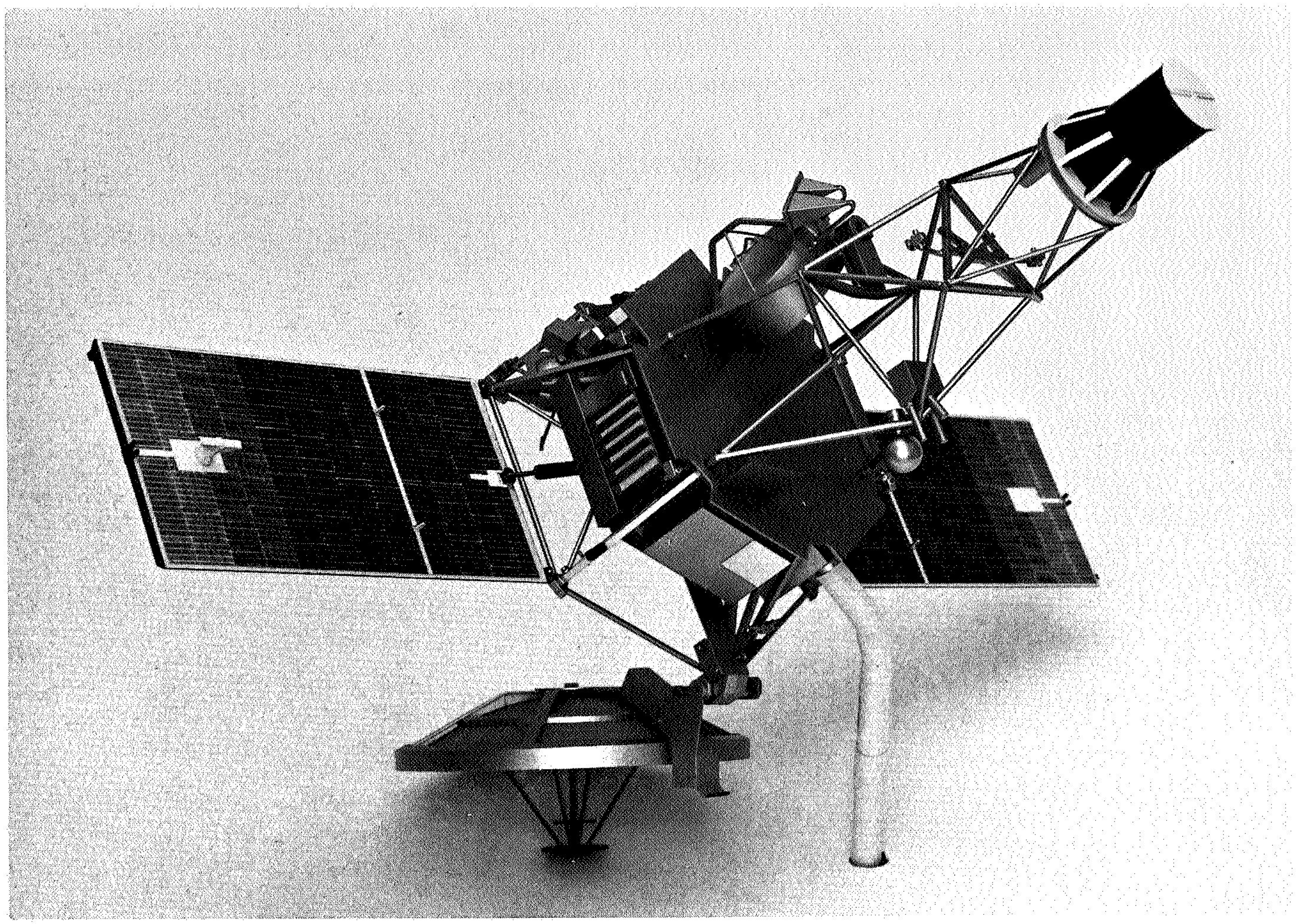


Fig. 2. *Mariner 1* spacecraft in the cruise configuration

this, the spacecraft maneuvers to acquire the Sun and the Earth, at which time it maintains its attitude for three months or so until it gets to Venus.

As Venus is approached, a radiometer scans back and forth until it locates the planet. Then it scans back and forth across the planet surface, making its measurements.

After hearing such a description, you might wonder how such a device was conceived and designed. There was no last year's model to redesign, for no one had ever done such a thing before. We had, from NASA, long-range plans, and a framework of desirable missions. We had a *Ranger* design which a year and a half ago was still untried. We had estimates of a booster capability. We had funds available to accomplish our task, and we had all the basic skills and experience that we had developed on related projects. With all these tools, we assembled a *study team* composed of senior technical people from all of the technical divisions (i.e., guidance and control, telecommunications, propulsion, space science, engineering mechanics, systems). The study team convened weekly to discuss the feasibility, in reasonable detail, of several missions. The results of the study by this team took the form of a proposal to the National Aeronautics and Space Administration, who make the basic decisions as to what missions are worthwhile. The recommendation that was made was that we could go to Venus in 1962, and make some meaningful scientific experiments. We proposed the use of a modified *Ranger* to save time. This idea was accepted by NASA and that was the start of the program.

To begin this program, we established a *preliminary design team* of senior technical people from all of the areas that would be represented on this spacecraft, very similar to the study team. We then established some mission objectives which were, basically, to further the

art of spacecraft development, and we established some design criteria and a definition of the spacecraft.

The total definition of the spacecraft at the start of preliminary design consisted of the following elements:

1. A pointable articulating system for planetary scientific measurements
2. Solar-power panels and power-storage and conversion equipment
3. Two-way communications equipment
4. Environmental control equipment
5. Modified *Ranger* attitude control based on tracking of the Sun and Earth
6. Midcourse maneuver
7. Instrumentation and data handling
8. Modified *Ranger* structure

This would be called the *Mariner* Venus prototype spacecraft. It would have an instrument to point at the planet for making discrete measurements of the planet Venus. Solar-power panels for electrical energy would be required. It would have a two-way communications system, through which we could receive data from the spacecraft and also send commands to it. It would have a modified *Ranger* attitude-control system, which really should be called an attitude-reference system, using the Sun and the Earth for attitude stabilization. It would have a midcourse maneuver for the vernier velocity correction, and it would have the instrumentation and data-handling capability on board to store up and send back information. In general, it was to be built around a modified *Ranger* spacecraft structure. Those were the elements needed to make up the spacecraft system. In attempting to combine these into a spacecraft, we invariably run into contests, for all these different elements are not automatically compatible.

The next step in the design process was to establish what we call competing characteristics. Any competing characteristics were resolved along some specified ground rules. The following is a list of competing characteristics for the *Mariner* Venus prototype:

1. Schedule
2. Compatibility of subsystems with each other
3. Reliability of subsystems to perform over the required lifetime in the following order:
   a. Injection-phase power system and spacecraft separation devices
   b. Tracking and trajectory determination up to midcourse
   c. Sun acquisition, pitch-yaw stabilization, and associated failure-detection telemetry
   d. Solar-power system
   e. Temperature control
   f. Antenna erection, Earth acquisition, roll stabilization
   g. Main telemetry system
   h. Midcourse maneuver
   i. Tracking and trajectory determination in the vicinity of the planet
   j. Scientific instruments and data-handling equipment
4. Contribution to technique for follow-on programs
5. Preflight operational simplicity after leaving the spacecraft hangar
6. Biological sterility

The first and most important consideration was the schedule. If we did not get this spacecraft together and off and on its way on time, it would not be worthwhile, so we compromised all sorts of things to meet the schedule. The second item was the compatibility of all the pieces with each other. The third consideration was the reliability of the subsystems to perform primarily in the order of flight sequence. Since this was to be a developmental shot, we wanted to prove out all the different facets of it, and we wanted each one to succeed. It was necessary for each one to succeed before we could pick up the next. These steps were in sequence. The first was to get through the injection phase, which is a period of severe vibration, atmospheric changes, and static acceleration. The next requirement was radio-tracking to determine the trajectory. In other words, once we had the spacecraft injected on its way, we had to find out where it was, what its velocity was, and what the midcourse requirements were. The next objectives were to get the Sun acquisition and all of the basic elements of the attitude-stabilization system in flight, and to get the solar-power system working.

All of the first five performances would occur perhaps within an hour or so after launching. About that time, then, if the temperature-control system has not been working properly, some components will start to overheat or freeze up. Temperature control thus becomes an important element as we get further along on the mission and get more information. Finally come the antenna erection, the Earth acquisition, and the roll stabilization. Then, for the next few days, the main telemetry system

starts telling us engineering information, coupled with a little science information, about this phase of the flight.

The midcourse maneuver performs the vernier trajectory correction. This complicated maneuver consists of sending radio commands to the spacecraft, making the propulsion maneuver, and reacquiring the Sun and Earth afterwards. Then the three- or fourth-month coast period will occur, at the end of which we will then analyze the tracking and trajectory determination in the vicinity of the planet to find out how good a shot was made. The final procedure is the achievement of the scientific measurements and the data handling for them.

Those factors, then, made up the basis for being able to proceed in a rational fashion in the preliminary design of the spacecraft. The team making up the preliminary design effort collected the requirements of each technical area. The leader of this team compiled all their requirements and tried to come up with what the spacecraft was going to be. The first action was to lay out a configuration, and to make some quick layouts on paper combining all these elements into a spacecraft. Something always goes wrong with the first of these configurations, necessitating going through reiterations. We thought we had a very good configuration and then we found out that there was something wrong with it, and when that factor was changed there was something wrong with something else. This procedure ended by tearing up the configuration and starting all over again. We went through perhaps a half dozen of these different configurations before we arrived at the *Mariner* prototype (Fig. 3). The technical representatives got together very often to resolve all the incompatibilities; for example, the solar panels that we had originally planned were not large enough to collect sufficient energy to satisfy the rest of the system. Two things could be done. One was to shrink the power demands of the other systems, or increase the size of the solar panels. The choice that we resolved among ourselves was to make the solar panels bigger, primarily because we could make them fit and we could solve all the rest of the problems that were associated with them.

Some other problems that we encountered, such as the trajectory requirements versus what we call the look-angle problem, will be discussed in succeeding lectures. The trajectory requirements which I am talking about here are primarily in the vicinity of the target planet. The particular arrangement of all the axes on the planetary horizontal platform to allow the instruments to see the planet is an example of a look-angle problem. This look-

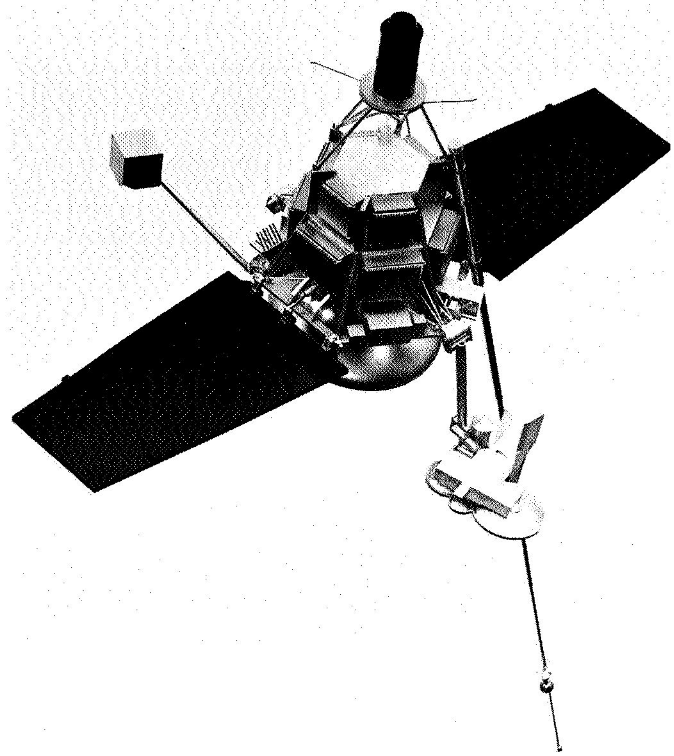


Fig. 3. *Mariner* prototype cruise configuration

angle problem, in even simpler terms, is trying to have all the elements which want to see in some direction, do so without any interference from the spacecraft.

Another problem was to try and satisfy all the requirements of the space scientists. On the *Mariner* prototype (Fig. 3) there are small box-like components scattered all over it, some on the end of long pipes, big ones, little ones, round ones, square ones, and all enclosing scientific instruments that were handed to us. There are four experiments on the *Mariner* prototype. The scientists concerned with magnetic measurements wanted a magnetometer to be 20 ft from the spacecraft. One is a radiometer for temperature measurements. One is an ultraviolet spectrometer for making spectral measurements in the Venusian atmosphere. The last was defined as particles and fields measurement. The particles and fields measurement resulted in 19 packages in all.

The finish of the preliminary design was summed up by considering what we had at the end of this time. Having gone through all the problems, we ended up with a configuration. We had a picture of how big the solar panels were and a picture of how big the telescoping

magnetometer boom was. We had some idea of the high-gain antenna and the whole arrangement of things. We had a flight sequence which all of the individuals concerned had been able to agree to in satisfying the basic requirements of the mission. We had a general plan for handling all the operational requirements for assembling the spacecraft, getting it checked out, put on the booster, and on its way.

About that time, the detail design period began, in which each division started working by itself on common ground rules. We had perhaps five or six different groups represented in the spacecraft during the preliminary design, all working together. Having defined a common working ground, the members of each group were able to go off on their own and do a fair job on their detail designs. The design at that stage was more difficult to change, but even so, there were always changes coming along. There was, of course, interaction among all the different areas because problems are never isolated into nice little compartments that can be solved alone.

One of the things that we ran into, for instance, on this design was in the area of temperature control. We found, initially, that on the upper electronic boxes there was too much electrical power dissipated as heat to be rejected from the available surface area. This caused a crisis because we really did not have any other place to put all this equipment. The solution was to take the high-powered material out and put it in small packages which we hung down along the bottom of the craft. That was one of the things that might happen very late in the design. We had the configuration all established, and suddenly there was a crisis which required a major change to the spacecraft.

About that time in the *Mariner* program, we had a design change when it became apparent that the launching of such a sophisticated prototype was not in the best interests of the program. The final *Mariner 1* was to weigh in the neighborhood of 450 lb. That was quite a small device when you think about it, so we looked at what we could take off. We took off most of the scientific components and other items here and there and ended up with about 500 lb. We then went back, essentially, to page one on the whole design, and said, "OK, if we were starting from scratch, what would we do?" Another study team was formed, and in about two weeks they came up with the conclusion that we could take a *Ranger* vehicle, modify it very slightly, use some of the *Mariner* prototype components, and put together something which had a small number of scientific instruments on it and which could go to Venus on an *Atlas-Agena* booster.

A proposal was then made to NASA offering this modification, *Mariner 1,* as an alternate approach to their requirement. Our proposal was accepted in a matter of a couple of days, and we started our preliminary design again. That was just a little less than a year before the shoot was to occur and there we were, back in the preliminary design. We had to work fast, and the entire preliminary design took two weeks, whereas the original one took three months.

To help design the superstructure, we built a little quarter-scale model out of balsa wood (Fig. 4). Looking at it very closely it can be seen that we sawed out different members because they did not work right. As we started bending the thing around and twisting it, we noticed that it had a peculiar bending mode that we had not analyzed, but it just did not look right. It happens, however, that this is the exact structure we are using right now, initially proved out on this little model. Our next procedure was to start in on our hard

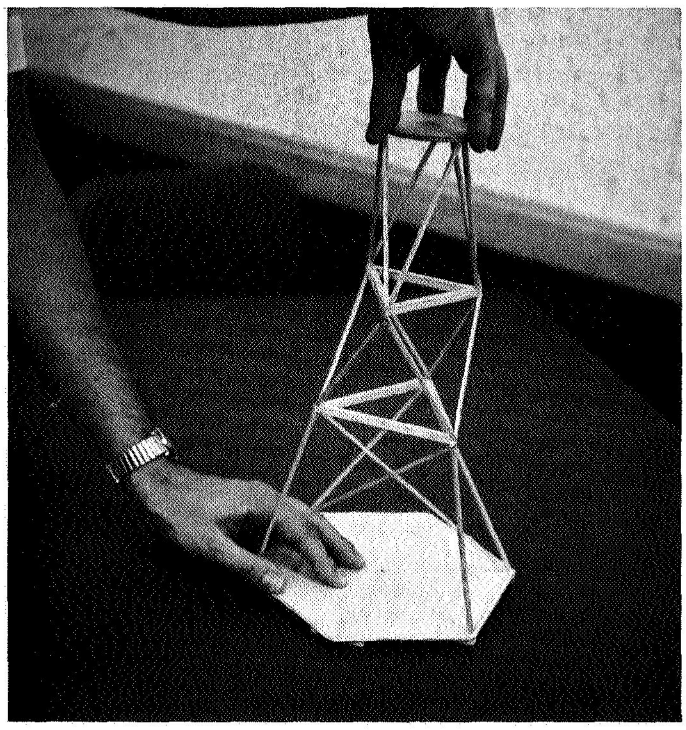


Fig. 4. Scale model of *Mariner 1* superstructure being tested

design, which was to design all of the structure. This work took about six weeks or so. All sorts of problems arose during preliminary design, but we managed to get quite a few scientific instruments on before we were finished. We took, for instance, a simplified version of the radiometer which had a single axis scan rather than the three degrees of freedom that the original one had, and that one item helped to solve our problem tremendously. We had an ion chamber which is the same device that we had originally. We had another component which was a contraction of the original particle and fields devices on the *Mariner* prototype into what was really necessary. The magnetometer which the scientists had insisted had to be 20 ft from the spacecraft on the *Mariner* prototype was now just bolted to it.

We had two major problems to solve as we went along into the design. One of them was that the *Ranger* had a nice structure already designed, making it very compatible with the *Agena* booster, but it had an Earth sensor on it designed for finding the Earth at lunar distances. Now we were contemplating a requirement for many, many times that distance, so another Earth sensor, which turned out to be twice the size and three times the weight, was needed. We did not think we had room to put it in, so we devised a scheme, a magician's approach, by doing it with mirrors. The problem was solved on paper by laying the Earth sensor on its side and putting in a mirror so that it just peeked around the rim of the antenna. And yet, it was no small feat getting it geometrically located. As we got into the design a little further, it became obvious that we were going to have to design that whole area again and we spent a major part of our design effort on that little corner of the spacecraft. The rest of it fell together quickly, but this area kept us busy.

The science integration problem was quite simple. We just offered to put on the craft what we thought we could, and the scientists generally agreed with it. We actually achieved more than the initial proposal had indicated that we could.

The design criteria were simple: putting something together and making it work. The competing characteristics in this particular case were all but non-existent. The whole design effort worked smoothly, primarily because all the people concerned were not shuffled around at all; exactly the same people who had done this sort of thing before were used.

The design of this vehicle is completed now, and undergoing final tests. We had some of the same problems that we had on the original, but having had practice before, we could go through all the different trade-offs and compromises that originally took months, in a matter of hours, because everybody was ready. For example, the solar panels were too small again, but we knew exactly what the solar panel people were going to say, we knew exactly what the telecommunications people were going to say, and we just went right down the line and we came up with the design very quickly. This was accomplished by virtue of having had practice in making an almost identical decision.

To further develop our subject, I would like to cover a few problems and our method of solving them. I did mention that to start a configuration, which is really a definition of the picture of the spacecraft as we see it, we collect inputs from each of the technical divisions. These technical divisions are the propulsion people, the space scientists, the guidance and control people, the telecommunications people, those concerned with structures, and others. Figure 5 illustrates hypothetical spacecraft designs as the different technical divisions would visualize them. It gives you an indication of what kind of task we have in bringing all of these fields together into one device. Obviously, these make many, many problems when the pieces are put together. They are primarily resolved by compromise. In other words, the power people would really like huge areas of solar panels which would turn out to be heavy and expensive, and so they fight to reduce the power demand. The space scientists really would like to sprinkle it with all sorts of instruments, all of them designed in the laboratory, but again under proper circumstances, they realize the difficulties and make their own compromises. This trade-off goes on continually during the preliminary design stage. Of course, as demonstrated with the *Mariner* prototype, there are a number of things to mount on a spacecraft. It always seems as though when we are trying to go through one of these configuration iterations there is always one more element to get on; i.e., everything can be put on, but one. We eventually did get them all on without undue compromise.

An approach toward solving this configuration problem is to take all the elements, once the spacecraft is defined in terms of discrete packages, try to arrange them functionally, and then see if they can be tied together with some sort of structure. This is an intriguing approach that takes many months. Another approach is to take an existing device of some sort and modify it. We started with a *Ranger* which you will recognize as about the lower 20 in. of the spacecraft (see Fig. 2). It has the

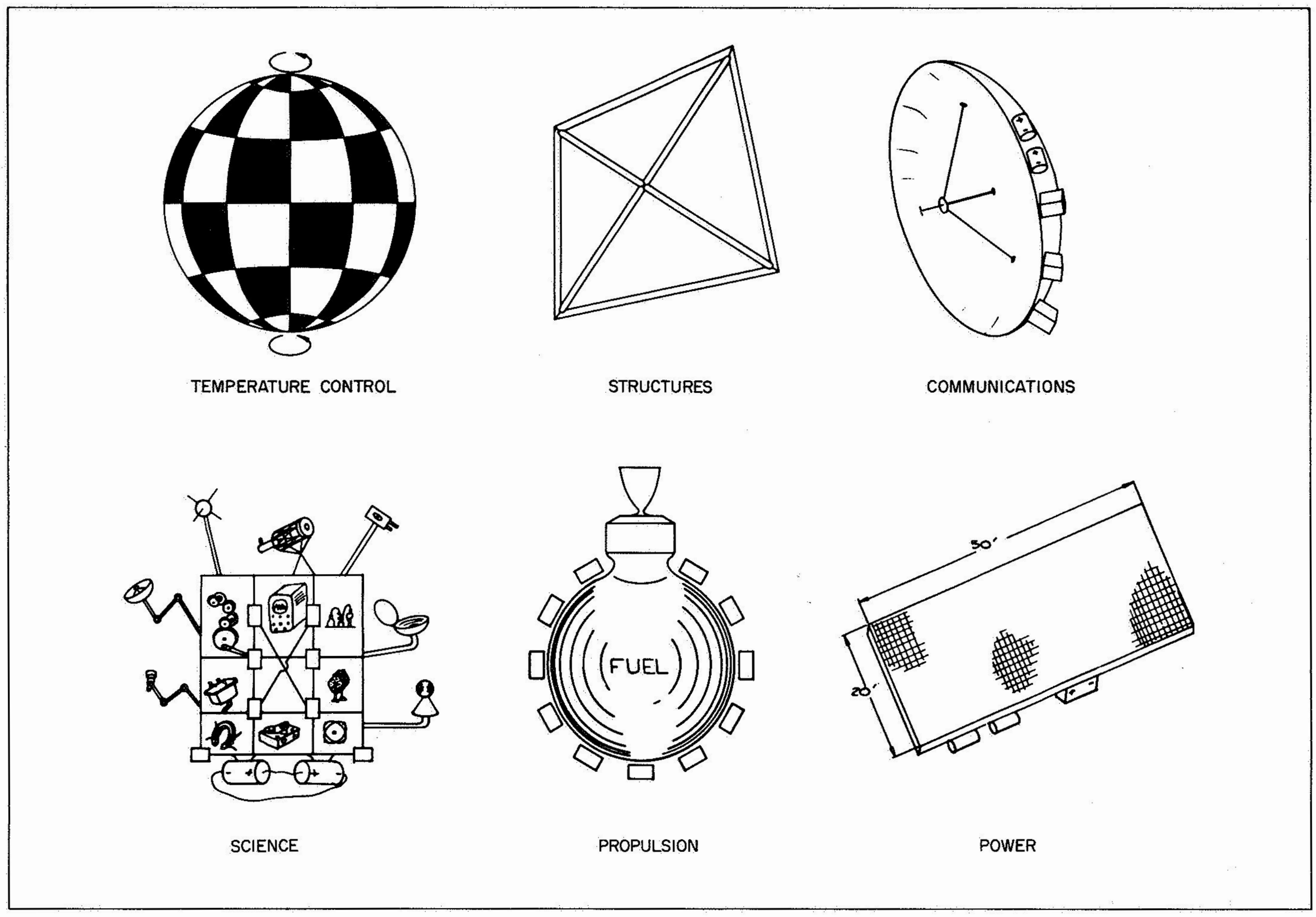


Fig. 5. Hypothetical spacecraft as designed by various disciplines

basic hexagonal shape, legs holding it up, boxes in it, and it has the 4-ft diameter antenna. No doubt, with that as a basic building block, despite its deficiencies, we saved ourselves a considerable amount of time, because we did not argue about what shape it should be, how it should be put together, or what the shape and size of these particular boxes would be. We just took the existing equipment and added new items.

Another situation that plagues us continuously is what I call fire-fighting: problems cropping up that we did not think of in the beginning. As the design progresses through the tests and the hardware development phase, new problems come up. These actually keep coming up all the way through to the time of the shoot. There is always something that has been forgotten. For example, we had on this particular program the problem of the little Earth sensor, which was in a convenient geometrical location, but in a terrible structural location. We designed the structure to support it and it looked good. We got through to the actual vibration testing of the spacecraft, for which we had assembled a complete prototype of this device, put it on an electromagnetic shaker, and had run it through some vibration levels at various frequencies. As we were about to complete those tests, we noticed that at one particular frequency we got a very high acceleration on that little sensor. It turned out to be an extremely sensitive instrument that did not even like to be left lying on the table without having trouble, much less being subjected to extreme vibration. We were faced then with the problem of how to remedy this particular problem. The over-all design was well along, and all the things were tied together. That sensor could not be picked up and moved to some other place on the spacecraft, because our earlier geometric exercise left no choice as to location. We tried fastening it down with a latch that would release it, and tieing it in very rigidly to the spacecraft. Another idea was to place some small,

simple dampers on it. The dampers were initially designed to be concentric tubes with a thin layer of grease in shear between them, which did a fine job of damping. That method worked, but we were still wary of it because it sounded very complex. After days of testing and some analysis and much ingenuity, we devised a scheme whereby we could make it work without keeping the Earth sensor permanently attached to the structure. This was done by putting a spring inside the damper, thereby allowing the damper to be pushed by the spring over half of the dynamic travel, and then become a damping member through the other half-cycle. This solved the problem of allowing the Earth sensor to fold away from the spacecraft without any interference. From there we went back into the design phase, trying to devise one to meet all the flight requirements as well. There were many pieces of pipe and cable and other parts which were previously located through which it had to thread. This particular problem is easy to describe because, so far as we were concerned, we found a successful solution for it. However, when that procedure was finished another aspect presented itself in that we discovered that the small spring which kept the damper pushed against the Earth sensor all the time, also tended, when we were separating from the booster, to push the antenna down somewhat. It pushed the antenna down, hit the booster, and that tipped the spacecraft. This was and is a very critical problem for this particular spacecraft. So we are now, at this late stage of the game, engaged in a program of analyzing the effects of this spring force—whether it really upsets the spacecraft, as we suspect it does, and if so, whether we should concoct a new approach, which in itself may create more problems. It is an iterative process which is sometimes difficult as we draw near to the time when we are supposed to launch the spacecraft.

By way of conclusion, I would like to present the characteristics that make up a designer. The first requisite is, of course, technical competence. You have to know your field or, in some cases, other fields, with few exceptions. The second requisite is to know the limitations of your abilities. It is more important that the area is covered than that you have tried to tackle something that looked like fun, but which you were not able to handle. Another essential quality in designing is creative ability. You have to be able to think of solutions faster than people can think of problems. You have to be able to take your ideas and twist them around into new shapes as required, for certainly the configuration is a good test of anyone's creative ability. You have to have an appreciation for the problems involving the system. You say, "Well, we would like to have this particular shape of a device here." But, we know that this proposal might interfere with someone else who has a valid point too, because he has his own problems. This really leads to the most outstanding quality of all, perhaps, which I call the ability to horse-trade in all kinds of areas, still realizing your own goals, without making any sacrifices that would hurt yourself, and at the same time contribute towards the system. This last characteristic, I think, is probably best summed up by calling it experience.

Scientific Instrument Design and Integration

BENJAMIN A. WAITMAN

Senior Development Engineer
Spacecraft Development Section

I. THE GENERAL PROBLEM

The problem of design and integration of scientific instruments for a space mission is a difficult one. It is difficult because the skills and responsibilities for accomplishing the task are usually split among many different technical disciplines. The purpose of this discussion is to examine this problem as an example of the complex modern engineering problem faced by the space industry today, and demonstrate to the student what these many skills are and how they interact.

The functions that need to be performed are:

1. Basic scientific objectives for space missions must be prepared and negotiated between the National Aeronautics and Space Administration and the scientific community.

2. These objectives must be translated into a specific group of instruments for any given flight.

3. Design and development of specific instruments must be accomplished by engineers.

4. In those cases where a controlling organization for a mission—such as the Jet Propulsion Laboratory for the unmanned portion of space exploration—must subcontract some of the scientific instruments, project and contract management functions must be performed.

5. Over-all design of the many types of scientific instruments and of the spacecraft must be accomplished in accordance with the mission schedule and must ensure that the most useful scientific data are obtained from a given flight.

To accomplish these functions, the following skills are required:

1. Scientific
   a. Understanding of basic and advanced physics
   b. Understanding of the total mission objective
   c. Special advanced instrument techniques
   d. Comprehensive data analysis

2. Packaging
   a. Mechanical design of electronic devices
   b. Testing of the instruments for environment
   c. Internal heat transfer analysis
   d. Fabrication and product management of mechanical assemblies
   e. Instrument weight control
   f. Project assistance on contracted instruments
   g. Design for environmental survival

3. Electronics
   a. Circuit design and development
   b. Data collection and storage
   c. Bench and ground-support testing techniques
   d. Servo theory application
   e. Power interface design techniques
   f. Instrument power control

4. Mechanisms
   a. Servo actuator design
   b. Non-servo actuator design
   c. General design of deployment devices
   d. Optical design techniques

5. Configuration and integration
   a. Look-angle analyses
   b. Structural and geometric integration
   c. Spacecraft weight, center of gravity, and moment of inertia control

6. Temperature control
   a. Heat-transfer mechanism interface analysis
   b. Power profile distribution control techniques
   c. Thermal operating limit analysis
   d. Heat-transfer instrumentation
   e. Simulation techniques for testing
   f. Surface treatment control and definition
   g. Design and development of temperature-control devices
7. Materials usage based on
   a. Knowledge of common and uncommon materials
   b. Thermal properties
   c. Structural properties under extreme as well as normal environments
   d. Optical properties
   e. Electrical properties
8. Mechanical and electrical technician support for assembly and testing
9. Project management and procurement skill which permit proper evaluation, selection, and direction of a subcontractor
10. Pyrotechnics support for unlatching of deployment devices and start of ignition for remote propulsion systems and actuation of certain types of deployment devices

The engineering process which implements these skills in performance of the basic functions described is shown schematically in Fig. 1.

Before flight system predesign can start we need three major independent areas of effort. One is the development of the science itself. You have all had courses at one time or another involved with the many disciplines of science. You have some feeling for the many years of effort that often go into developing a technique that is to be used for observing various types of phenomena. Individuals such as Dr. G. Beadle and Dr. J. Lederberg in Biology have spent years looking at the life cycle of a single cell or a group of cells. They have developed techniques that will be useful when trying to determine the presence of life on remote planets. Dr. G. Sonnet of the National Aeronautics and Space Administration similarly has developed a specialty in magnetics. His ideas, in the

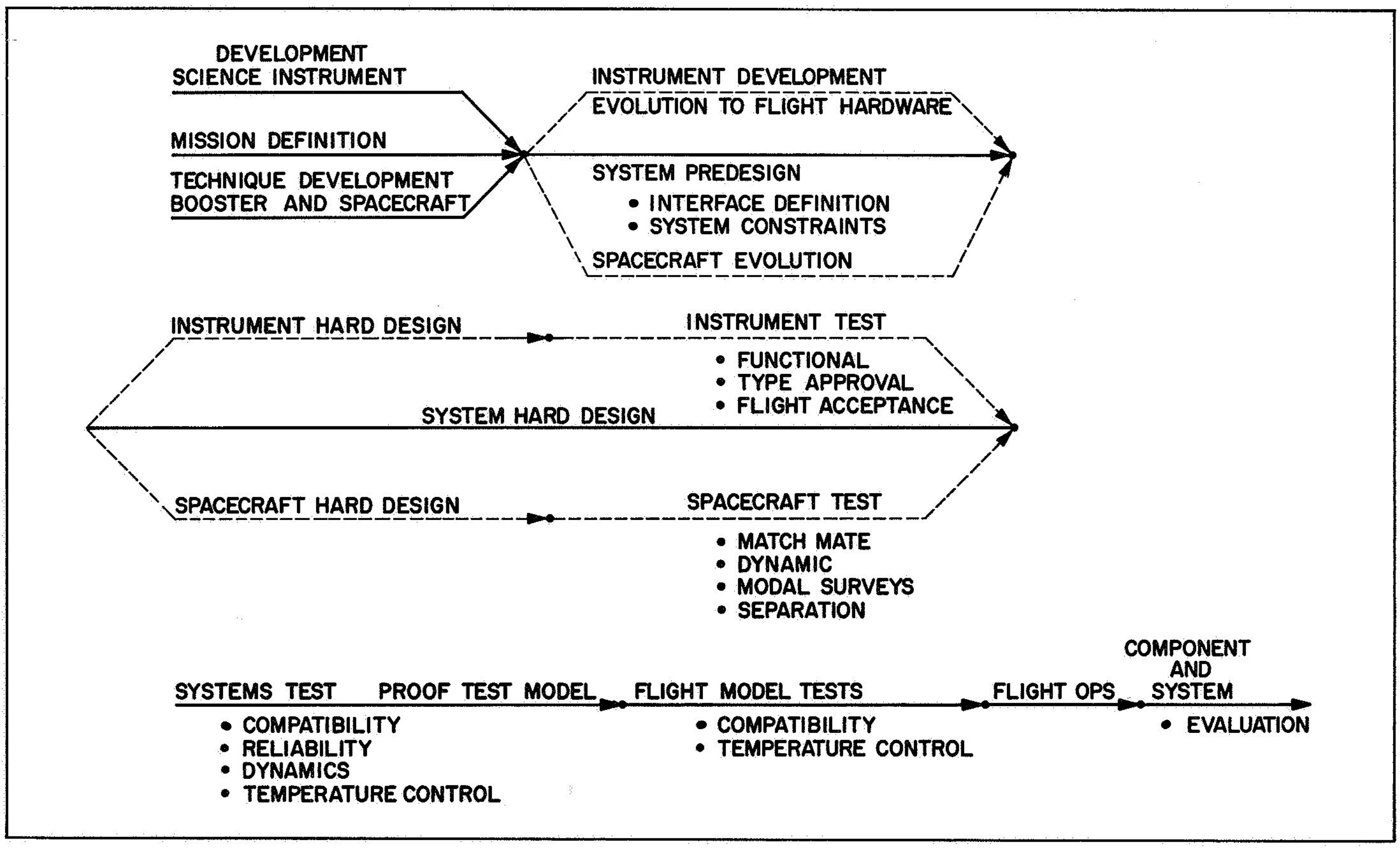


Fig. 1. Spacecraft/scientific instrument development sequence

form of actual instruments, are flying today to discover more about the universe and our solar system in particular. Various scientists at Mt. Wilson, Palomar, and other observatories have been viewing extraterrestrial bodies for years in hopes of learning more about the universe around us. Today they have the opportunity to probe the universe directly.

Secondly, the "hardware" man just recently has learned how to harness enough energy to counteract the Earth's gravity field. Therefore, he has been going through a period of developing booster and spacecraft design techniques.

These two areas of effort combine in a third, which is described as mission definition. Missions are defined jointly by NASA and the scientific community as to what is desired, compromised with what can be done with the existing state of the art. Several difficult questions must be answered during this period. Some of these are:

1. Will the spacecraft be attitude-stabilized in trajectory?
2. Are long flight times with high information-transmitting-rate capabilities available or required?
3. Will data storage be required?
4. Are large changes in distance from the Sun involved?
5. Do we want to near-miss another body, impact it, or go into orbit around it?
6. Is the spacecraft a satellite or deep-space probe?
7. How important is schedule?

Following resolution of these and many more questions, a specific group of instruments is selected for a flight and the preliminary system design can begin. Preliminary design is the major period of system formulation during which all functions of the spacecraft and its equipment are defined, and their interfaces specified. All of the basic compromises between system desires and capabilities are resolved during this period.

Following predesign, "hard" design for fabrication is started. Effectively, the spacecraft and its equipment can now be developed independently, but, of course, problems arise which must be solved. Scientific instrument integration starts during predesign and should finish at the end of predesign. In practice it often carries over for some time into hard design in the form of support to the scientists. The end product of integration is a specification control drawing which defines complete mechanical and electrical interfaces for a device.

The remainder of Fig. 1 is relatively self-explanatory. All of the enumerated skills are employed throughout the entire process. Additional skills not mentioned, but of extreme importance due to the complexity of the problem, are tact, clarity of technical expression by the written and spoken word, and common sense.

In what ways are the problems of spacecraft instrument design and integration so complex? Succeeding sections of this discussion will attempt to demonstrate the answer to this question. We will cover one example ("look angles") in detail as a demonstration of a problem that on the surface appears to be simple and amenable to straightforward analytical or graphical attack. As will be seen, even the simple-appearing problem can become complex. Several other examples will be covered generally in the text with amplifying discussion during presentation.

Astronomers over the years have determined, primarily by observation, the motion of the various heavenly bodies with respect to one another. The basic laws of their motion are thought to be well understood, but such is not the case. Kepler's and Newton's equations of motion are correct with respect to two bodies in an isolated system, but to date no one can solve the equations for three or more bodies exactly in closed form. When we launch a spacecraft, another body is added to an n-body problem. The motion of the spacecraft is approximately in a plane in space, but what plane? The plane of motion is difficult to determine, and requires at least three space coordinates to define it with respect to some known reference. Therefore, our geometry problem is a three-dimensional problem of the motion of a spacecraft in a dynamic gravitational field of n bodies which, when solved approximately, is the trajectory.

When designing a device such as a spacecraft, we engineers use the normal orthogonal projection scheme. The device has physical size. Therefore, when moving in space the device would have six degrees of freedom: three position degrees for its center of mass, and three rotational degrees about its center of mass. Now, the physics of the universe is very orderly. The scientists have, in general, hypothesized their theories of physical phenomena with respect to this orderliness. The devices which they want to fly contain sensors to investigate the various phenomena. In general, these sensors are highly

directional. When we install these devices on a spacecraft, it is required that we position these sensors with respect to the spacecraft and the orderliness of the universe. It would be useless to have a spacecraft tumbling aimlessly through space; the output of the sensors would be meaningless. Similarly, it does no good to have these sensors look at the spacecraft. Therefore, a geometry problem that is unusual presents itself. As engineers, we conceive hardware in two dimensions on paper. Perhaps we draw an isometric of our hardware, but in general we just build it and see if everything fits. Here we are presented with a problem which is three-dimensional to begin with. How do we translate normal "two-dimensional" design into a situation where three dimensions is the starting point? Our thinking must be adjusted.

The solution of this geometry problem forced the development of a design technique which we call "look angle" analysis. Discussion of this technique in detail follows in Section II.

II. LOOK ANGLES—AN EXAMPLE OF A COMPLEX INTEGRATION PROBLEM

A. General

One of the more difficult problems encountered when performing scientific instrument integration is a problem in geometry, or what we call look angles.

It will be recalled that the devices of the scientists must be installed with respect to the spacecraft and with respect to the orderliness of the universe. The point-to-point motion of the spacecraft is also orderly, whether or not we can solve its equations of motion exactly. Therefore, the first step of look-angle analysis is a knowledge of the trajectory of the spacecraft; what is its translational path in space coordinates. With this knowledge we tie down three of the spacecraft's six degrees of freedom as functions of time. Secondly, we need to tie down the three rotational degrees of freedom as functions of time with respect to the trajectory and the three-dimensional geometry of the actual spacecraft. This is called attitude control. Since the analytic trajectory coordinate solution is solved in discrete mission phases, the analytic solution will be discontinuous between phases. (What these phases are and how they are determined will be covered later.) However, the actual attitude of the spacecraft and its actual trajectory are continuous with time. Therefore, a second set of controls—called local guidance coordinates—which are continuous throughout the flight is necessary. These two sets of coordinate controls coupled with the spacecraft's actual geometry of construction allow the solution of the look angle from any point on the spacecraft.

Therefore, a look angle defines what can be seen, or what is the field of view for a sensor from any desired point on the spacecraft without intervening structure.

Look angles are separated into two types: fixed look angles—those fields of view of sensors fixed to the spacecraft, or having a fixed relationship to the attitude and body coordinates of the spacecraft; and moving look angles—those fields of view of sensors which move with respect to the attitude and body coordinates of the spacecraft during operation of the sensor. (In the general sense, all look angles are fixed fields of view in which either a fixed or moving sensor with its field of view

operates without intervening structure. For our purposes here, however, we will treat the field of view of the sensor as the look angle which is fixed or moving.)

B. Spacecraft Fixed Look Angles

In Fig. 2 we see that the fixed look angles are defined by direction and data acceptance fields of view. When referring to direction, a specific nomenclature is defined as follows:

1. Respect to the Sun
   a. To the Sun at 0 deg
   b. Away from the Sun at 180 deg
   c. At an angle to the Sun 0 to 90 deg
   d. At an angle away from the Sun 90 to 180 deg
2. Respect to the ecliptic
   a. In the ecliptic plane or in a plane parallel to the ecliptic plane
   b. In a plane perpendicular to the ecliptic plane
3. Respect to the trajectory
   a. In the direction of the trajectory
   b. In advance of the spacecraft's motion
   c. In reverse of the spacecraft's motion
   d. In the opposite direction to trajectory
   e. In advance of the Earth's orbital path
   f. In reverse of the Earth's orbital path

DIRECTION
- RESPECT TO SUN
- RESPECT TO ECLIPTIC PLANE
- RESPECT TO TRAJECTORY

DATA ACCEPTANCE FIELDS OF VIEW
- CIRCULAR CONES
- HEMISPHERES
- SPHERES
- PYRAMIDS
- FANS
- CYLINDERS

Fig. 2. Body-fixed look angles

Some of this nomenclature seems redundant, and it is, but these phrases are all used by the scientist. The attitude of a spacecraft, when functioning properly, will be known during the entire flight. Therefore, combinations of these nomenclature uniquely define a direction for a sensor for the entire flight. The directions for the many sensors are specified by the scientists involved with the various experiments, based on the data desired when studying specific scientific phenomena.

Each of these sensors has a data acceptance field of view which can be of any geometric volume (Fig. 2). Each of the sensors is so positioned that in its desired direction its data acceptance field of view is not obscured by other spacecraft structure.

C. Moving Look Angles

Moving look angles in general result from tracking celestial bodies which move with respect to the spacecraft. To introduce the concepts involved with moving look angles, we must go back to the basic definition of a trajectory and its basic coordinates definition. Using this definition, we must then relate guidance coordinates, referenced to the spacecraft and to the trajectory. Using the guidance coordinates definition, we must progress to the spacecraft's geometry coordinates and the definition of moving fields of view which are not obscured by intervening structure.

1. Definition of the JPL/STL Standard Space Trajectory Coordinate System

Following is a definition of the Standard Space Trajectory Coordinate System used and mutually developed by JPL and Space Technology Laboratories. The listed coordinates are mathematically defined and interrelated and are the basic inputs and outputs for a program used in an IBM 7090 complex.

a. Primary system

The primary system is "space fixed" (Fig. 3). It is a standard "right-hand" coordinate system. There is a secondary system which is always referenced to the Earth. The effects of the Earth's rotation enter as vector cross products and the conversion between primary and secondary systems, which is programmed for an IBM 7090, is quite involved. Since the secondary system is not too frequently used in the output/input format, it will not be treated in this simplified discussion.

The location of a probe and its motion are usually defined in either a set of Cartesian or Spherical coordinates. This is true for the primary system under discussion.

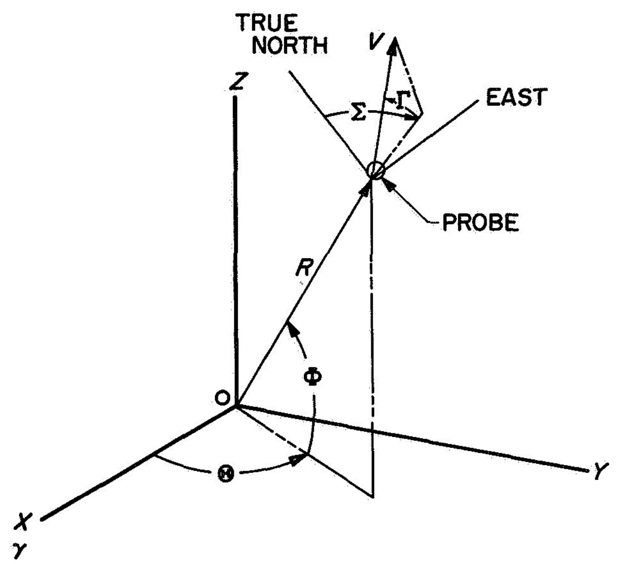


Fig. 3. Space-fixed trajectory coordinates

In Cartesian coordinates, the definition is a relatively simple matter of selecting the direction of two major axes and the remainder of the system falls almost automatically into line. In Cartesian coordinates, we have the standard X, Y, Z, T coordinates (T, a synthetic parameter, is the time after time of launch or other suitable time reference) which define the location of the spacecraft with respect to the origin of the system. Since a spacecraft's motion or its velocity is the time rate of change of position, we have additionally $\dot{X}$, $\dot{Y}$, $\dot{Z}$ (the dot referring to change in a coordinate with respect to time), which are the Cartesian components of the spacecraft's space-fixed velocity vector V. Therefore, the Cartesian set of coordinates is X, Y, Z, $\dot{X}$ $\dot{Y}$, $\dot{Z}$ T.

This set of coordinates is only useful if it is fixed in space or its motion clearly defined with respect to some known reference. The JPL/STL Cartesian coordinates are located by pointing the X-axis in the direction of the mean Vernal Equinox (First Point of Aries, γ) of the date in question. The Vernal Equinox is that point in space through which the Sun appears to pass going South to North, directly over the Earth's equator. The point is established during the day, March 21. The Vernal Equinox is not fixed in space, but precesses with the Earth at about one revolution every 25,800 years. A second axis Z is aligned to the Earth's mean spin axis of date. Therefore, the X-Y plane becomes the mean equatorial plane of date. How this set of coordinates is used to proceed from the Earth to another point in space will be described later. However, the coordinate system is defined initially with respect to the Earth.

The set of Spherical coordinates uses the same direction definitions as the X and Z axes of the Cartesian. However, position is now defined by the radius R from the origin to the spacecraft and two angles Θ, which is the Right Ascension of the spacecraft, and Φ, the Declination of the spacecraft. Θ is the angle measured from the origin-Vernal Equinox line in the mean equatorial plane, counterclockwise to the projection of R into the equatorial plane. Φ is the angle measured from the same projection of R to R in the plane of R and its projection. Φ corresponds to latitude of the usual Earth coordinates, latitude and longitude.

The motion of the spacecraft is defined by the absolute velocity vector V in Spherical coordinates. Since V is not necessarily in the direction of R, a separate "right-hand" set of axes is required to relate V to R and other coordinates.

This separate set of axes is defined by a plane, called the Local Horizontal (Normal) Plane, the direction of R, and the direction of True North. The Local Horizontal Plane is defined perpendicular to the direction of R at the probe. The direction of True North originates at the probe and extends in the direction of Z, but in the Local Horizontal Plane. Hence, the True North axis intersects the Z axis. The Local Horizontal Plane is, therefore, a plane tangent to the sphere generated by the loci of all positions of the probe at radius R from the origin. The third axis is East and is in the plane of the Local Horizontal. To treat V in this separate set of axes, V is analogous to R; therefore, two angles are additionally required to define the direction of V. These angles are Σ, the Azimuth of the projection of V into the Local Horizontal Plane measured clockwise from the direction of True North to the projection of V, and Γ, the Pitch (Path) angle, the angle measured from the projection of V in the Local Horizontal Plane to the vector V in the plane of V and its projection in the Local Horizontal Plane. Therefore, the set of Spherical coordinates is R, Θ, Φ, V, Σ, Γ, T. The time is the same for both sets, Cartesian and Spherical.

Figure 3 is a schematic of the coordinate system defined above. Figure 4 is a list of mathematical expressions relating the Cartesian and Spherical coordinates.

$$R^2 = X^2 + Y^2 + Z^2$$

$$V^2 = \dot{X}^2 + \dot{Y}^2 + \dot{Z}^2$$

$$\tan \Theta = Y/X$$

$$\tan \Phi = Z/\sqrt{X^2 + Y^2}$$

$$\sin \Phi = \frac{Z}{R}$$

$$\cos \Phi = \frac{\sqrt{X^2 + Y^2}}{R}$$

$$\dot{X} = V\,(\cos \Sigma \cos \Gamma \sin \Phi \cos \Theta - \sin \Gamma \cos \Phi \cos \Theta - \cos \Gamma \sin \Sigma \sin \Theta)$$

$$\dot{Y} = V\,(\sin \Theta \sin \Gamma \cos \Phi - \cos \Theta \cos \Gamma \sin \Sigma - \cos \Sigma \cos \Gamma \sin \Phi \sin \Theta)$$

$$\dot{Z} = V\,(\sin \Gamma \sin \Phi + \cos \Gamma \cos \Sigma \cos \Phi)$$

Fig. 4. Mathematical relationships between the Cartesian and Spherical coordinates of the JPL/STL Primary Space Trajectory Coordinate System

b. Phases

The use of the primary coordinate system described above is stipulated by a specific mission, whether Earth satellite, lunar probe, interplanetary probe, or deep-space probe. The use is broken up into three phases if three are necessary.

Phase 1 is always Earth-centered (Geocentric) regardless of the mission contemplated. The origin of the coordinate system is at the center of the Earth with the axes defined in the preceding.

Phase 2 trajectories are of two types, lunar or interplanetary. For a lunar mission, Phase 2, the origin of the coordinate system is the center of the Moon. The X and Z axes are referenced as in Phase 1. Phase 2, interplanetary, is Sun-centered (Heliocentric). The coordinates are referenced and transferred as follows. At the completion of Phase 1, the Phase 1 coordinate reference undergoes two rotations and one translation in space. The first rotation is of the X axis (origin, Vernal Equinox line). This line is rotated to the direction of the mean Vernal Equinox of 1950. This date has been established by international agreement. The second rotation is the rotation about the X-axis of the plane X-Y, the mean equatorial plane of date, into the plane of the ecliptic, the common plane of space motion of the Sun and planets such as Earth, Mars, Venus, etc. After these two rotations, the entire coordinate system is linearly translated to the center of the Sun. Figure 5 shows these rotations and translation.

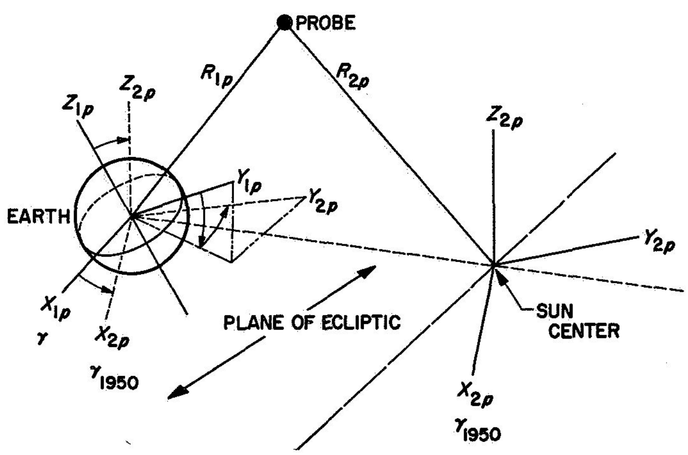


Fig. 5. Transition from Phase 1 (1p) to Phase 2 (2p) interplanetary

Phase 3, interplanetary (there is no Phase 3 for lunar missions) coordinate reference is established at the target, Mars or Venus, for example. The first step in shifting coordinates is a linear translation of the axes as defined for Phase 2, interplanetary, to the center of the target. Then, depending on the option desired, the ecliptic plane of the Heliocentric system remains fixed or is rotated to the mean equatorial plane of the target. For example, if Venus were the target, the ecliptic plane would most probably be used since there is no known axis of spin and equatorial plane for Venus. However, for Mars, a rotation of the ecliptic plane into the Mars mean equatorial plane would probably be used since Mars does rotate on its own axis. For transferral from Phase 2 to Phase 3, interplanetary, however, there is no change in the direction of the X axis, the origin Vernal Equinox line of 1950.

The time T after launch for the shifts between phases is a function of the mission, and is defined from the predominant gravitational field affecting the equations of motion of the spacecraft. For example, the gravitational effects of the Earth-Moon combination predominate for about five to eight days; therefore, interplanetary Phase 1 lasts this long. The Sun's gravity field dominates the motion during Phase 2. Phase 3 starts at a previously stipulated distance from the target, presently approximately 0.01 Astronomical Units. One Astronomical Unit (AU) is the mean distance from Earth to Sun center, approximately 93 x 10^6 miles.

2. Local Space Guidance Coordinates

A spacecraft's operation does not suffer from discontinuities due to limits of analytic capability. Therefore, a set of guidance coordinates that is continuous for the entire flight is needed. Such a local spacecraft or probe system of guidance coordinates has been defined by JPL (Fig. 6). The system is defined by one axis and two planes. The main axis is the line from the probe to the center of the Sun. One of the planes is perpendicular to this axis at the probe. A second axis is defined by a second defined plane and its intersection with the first plane. The second plane is arbitrarily defined by the probe, the Sun, and the star Canopus, which is approximately 75 deg below the plane of the ecliptic. The third axis of a "right-hand" system is then perpendicular to both the prime axes, probe to Sun, and the plane intersection described above.

The first plane, perpendicular to the primary axis, probe-Sun line, would be in the minus R direction for Interplanetary Phase 2. The local coordinates described here, however, are the same for all trajectory phases. The IBM 7090 computation program contains all the required conversion subroutines for going from these local coordinates to the trajectory coordinates described in Section II C-1.

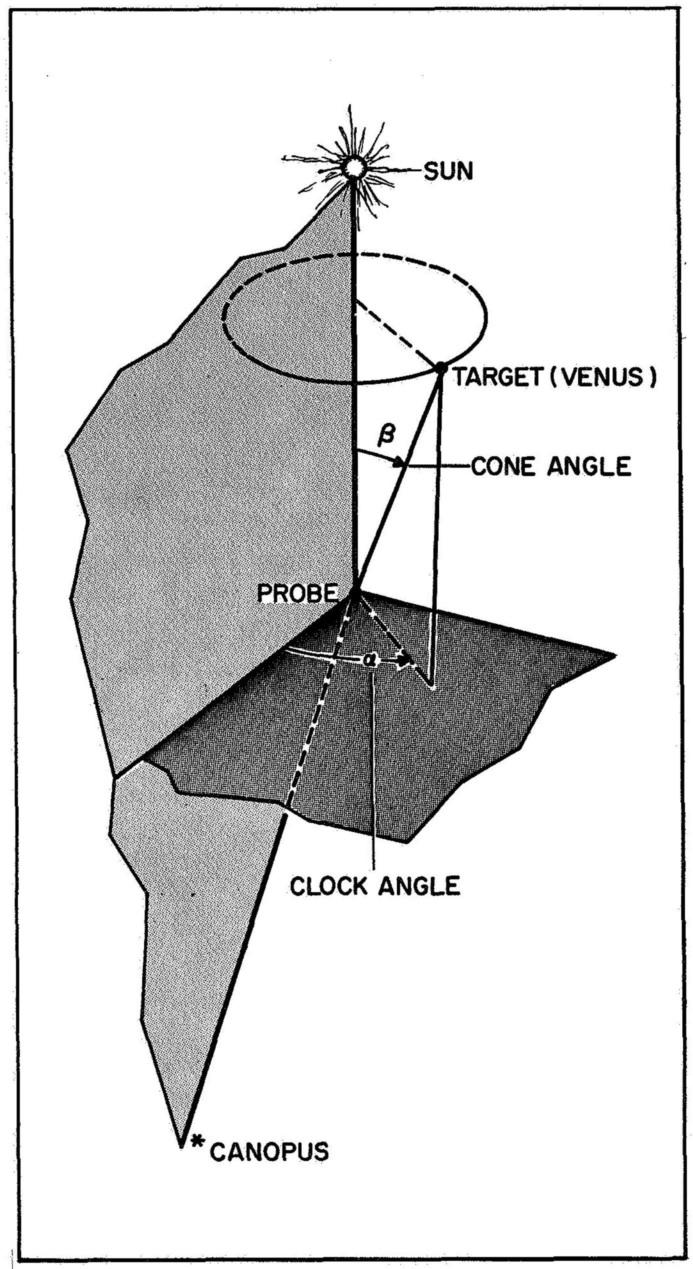


Fig. 6. Local space guidance coordinates

In local coordinates, the position of a target, Venus or Mars, is defined by reference to the probe by two angles, $+\alpha$ the "clock angle," and $+\beta$, the "cone angle."

β is the positive angle measured from the probe-Sun line to the probe-target line. Positive α is measured in the plane normal to the probe-Sun line from the intersection of this plane and the plane of Sun-probe-Canopus, counterclockwise to the line of projection of the probe-target line in the normal plane. The direction from the probe to the Sun is positive. There is at present no necessity of positive/negative directions for the remaining axes.

3. Relation of Trajectory Coordinates and Local Space Guidance to a Spacecraft Coordinate System

We must now relate the spacecraft's geometry to the trajectory and the local space guidance coordinates (Fig. 7). A base plane is defined. For the *Mariner* this base plane is the ecliptic plane. The roll axis of the vehicle was arbitrarily established so that the Y-axis of the X-Y-Z spacecraft coordinate system was in the base plane. The roll axis is pointed at the Sun. Both α the clock angle, and β, the cone angle, can now be inserted into this spacecraft coordinate system. β is the angle between Sun-spacecraft-target planet. β is composed of two spacecraft-defined component angles, hinge angle and swivel angle, which apply to the device that will track the target. Hinge angle is the angle measured from the spacecraft-Sun line in the base plane to the projection of the spacecraft-planet line in the base plane. Swivel angle is measured from the spacecraft-planet line projection in the base plane to the spacecraft-planet line.

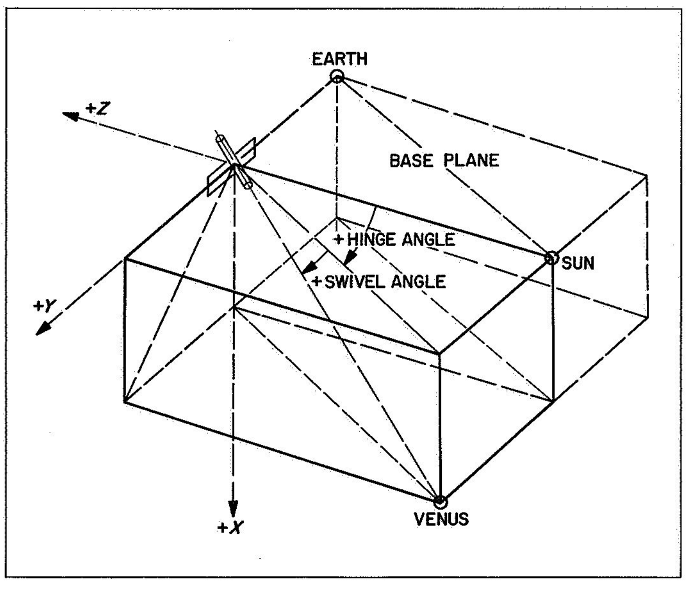


Fig. 7. Definition of hinge and swivel angles (cone and clock angles omitted for clarity)

4. Design Use of Moving Look Angles

Use of the point-to-point trajectory coordinates in space for a point mass generates a table of computer output data in the seven basic parameters discussed in Section II C-1. These data are computer-converted to the local spacecraft guidance coordinates that are applicable for the entire flight for the celestial bodies to be tracked. Imagine that you are riding on the spacecraft. From your position on the vehicle it is easily seen that if you look at the Sun you see one half of a celestial sphere with the spacecraft at the origin. Should you look directly away from the Sun you see the other half of the same celestial sphere. Using these two reference directions it can be seen that if there is a target in your field of view, the angle from the Sun to you on the spacecraft, to the target is the angle β. The angle measured between the spacecraft-target line and the spacecraft-Canopus line projected into a plane perpendicular to the reference directions at the spacecraft is the clock angle α. Therefore, a plot of the cone angle and clock angle as a function of time along the trajectory in polar coordinates can be made. Figures 8 and 9 are such plots for a *Mariner* spacecraft-Venus encounter. Radial distance is cone angle; zero through 360 deg around the periphery is the clock angle. The motion of Venus is superimposed as shown. The shaded area gives the variation in planet diameter as a function of time. The numbers along the planet motion lines are times from launch.

Now, it turned out that to satisfactorily track Venus for one of the *Mariner* configurations, a boom was required. The boom was used to establish a plane perpendicular to the spacecraft-Venus radius. This plane is the Local

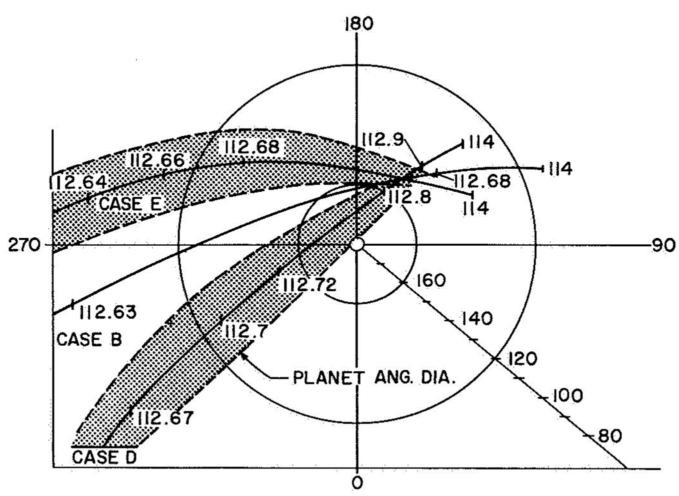


Fig. 8. Venus trajectory—112-day Venus projection

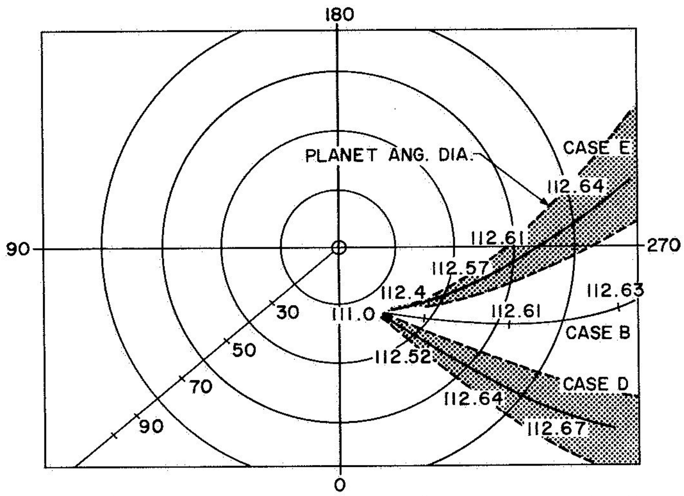


Fig. 9. Venus trajectory—112-day Venus projection

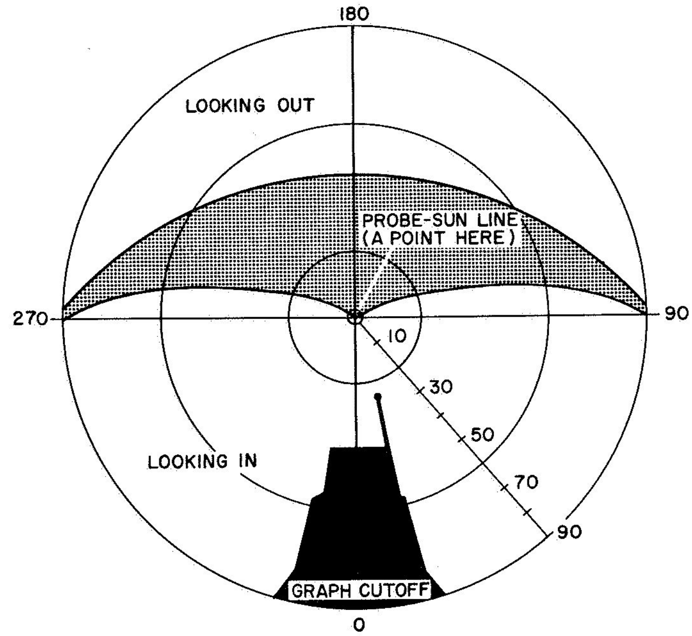


Fig. 10. *Mariner* PHP shadow graph

Horizontal plane in trajectory Phase 3 and was designated the Planet Horizontal Platform. The direction of this radius from the Horizontal Plane was plotted as a function of cone and clock angles with respect to the spacecraft to determine the interference with spacecraft structure. Superimposed on the direction during this process was the data acceptance field of view for the tracking devices. At the time, we were not sure whether we would be "looking out" or "looking in" with the instruments. These directions mean from the Horizontal Plane away from the vehicle or from the plane toward the vehicle with hinge angle at 105 deg. Figure 10 shows one of the two shadow graphs. The coordinates are again cone and clock angles and there is another half to this celestial sphere as before. Now if you superimpose Fig. 9 in which Canopus is fixed, and Fig. 10 in which the boom is fixed, you get a direct determination of where to put the Canopus tracker and the planet-tracking devices with respect to each other and the spacecraft roll axis. A roll of Fig. 10 with respect to Fig. 9, until the planet can be seen at all times, defined Canopus zero position with respect to the spacecraft. Therefore, roll variation throughout the flight is now defined.

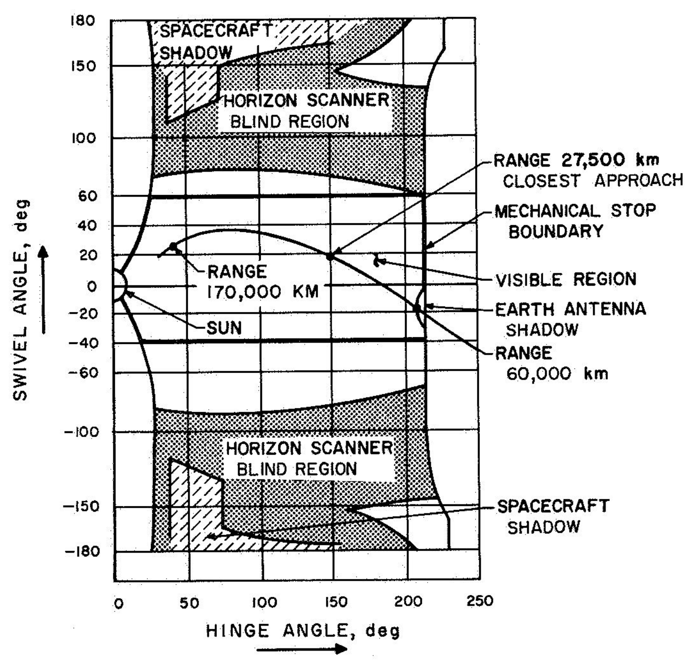


Fig. 11. Spacecraft shadows, and hinge and swivel angles

With the knowledge of cone and clock angle, and the spacecraft's roll attitude defined, the data may be converted into a spacecraft hardware drawing showing shadows, and hinge and swivel angles. Figure 11 is such a plot. This plot determines directly the available data acceptance fields of view for the scientific instruments on the horizontal platform. Trajectory information is added and, should scan over the target surface be desired, the extent of scan can be superimposed to determine if the available field of view is acceptable.

III. EXAMPLES OF MECHANICAL DESIGN PROBLEMS

A. Electronic Packaging

Figure 12 is an example of what we call a standard electronic module. As you can see, the assembly is quite complex. The design is based on a dual-functioning mechanical structure that helps support the spacecraft as well as provide mounting surfaces for the circuit boards. These circuit boards are phenolic in which either copper or silver is laid for the basic wiring together of the components. The wiring you see connects discrete subassemblies to their common interconnect with the remaining spacecraft wiring system. Each terminal has a specified spacing which is checked. All soldered connections and other details are checked by ten-power microscopes to verify proper fabrication. There are several hundred of these types of modules in a typical spacecraft.

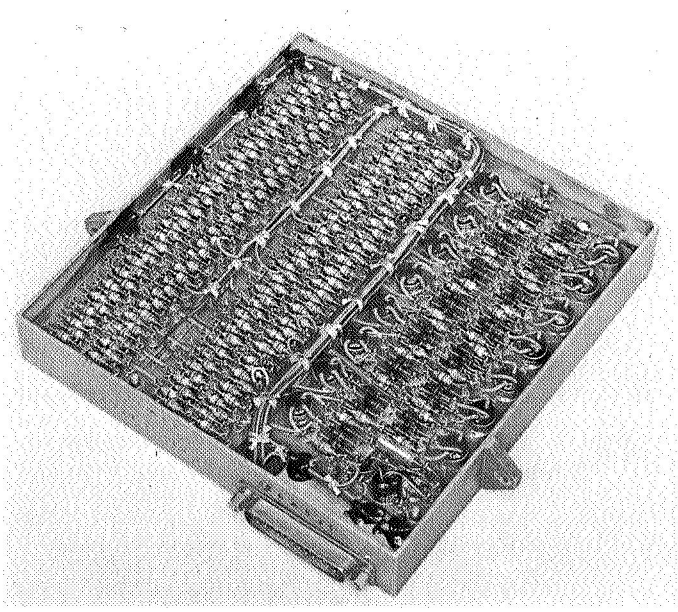


Fig. 12. A typical electronic module

Figure 13 is a typical breadboard of a Central Control and Sequencer (CC&S), the electronic "brain" for a spacecraft. This installation is about 3 ft wide and 4 ft high. It has a few extra testing circuits and is obviously quite heavy. The problem is to keep all of the basic equipment and lighten it for use in a spacecraft. Figure 14 is the result. The whole box is 12 × 15 in. The CC&S is now in the seven 6 × 6-in. modules in the upper left corner of the box. This compaction and reduction of weight is about 90% a mechanical design problem and 10% electronic. This assembly would now be functionally

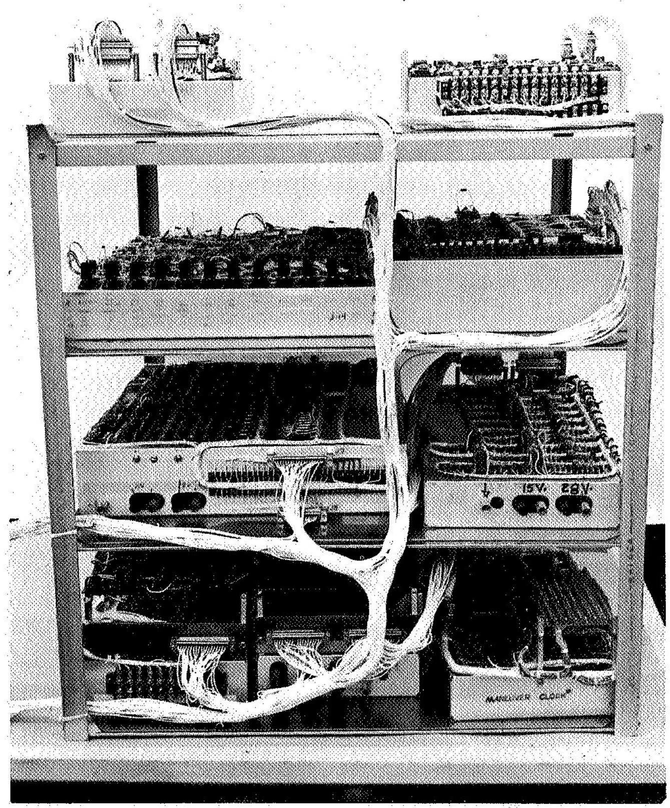


Fig. 13. The CC&S breadboard before flight packaging

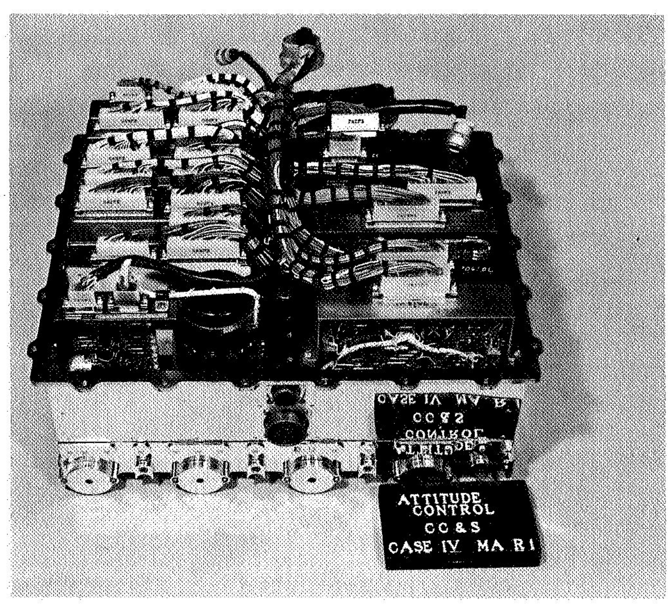


Fig. 14. The CC&S after flight packaging

tested and type-approval environment tested. Should there be a failure it would be found, evaluated, corrected and tested again. This process is kept up until failures do not occur in test and hopefully will not occur in flight.

B. Microwave Radiometer

Figures 15, 16, 17, 18, and 19 show various aspects of a *Mariner* spacecraft microwave radiometer. This device is to be flown past Venus to determine the temperature profile of the Cytherean atmosphere. It receives radiation at 15, 22, 35, and 75 kilomegacycles. You will notice the complicated structural design. The system requires the routing and tie-down of many links of microwave wave guide. The four horn-like protrusions look at space so that the 3°K can be a known reference. The main objects of interest here are the single-piece chassis and the noise-generator coupler as examples of the complex design required.

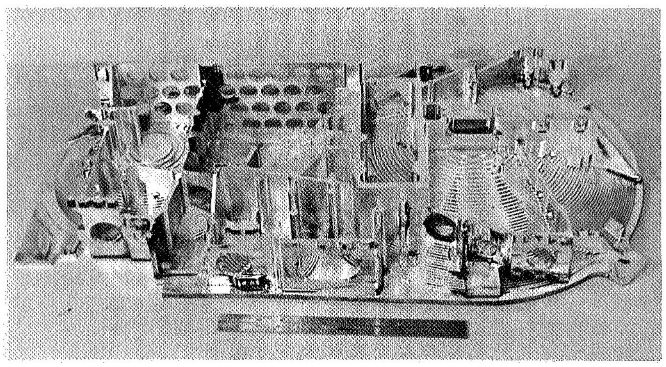


Fig. 15. Back of radiometer chassis during machining

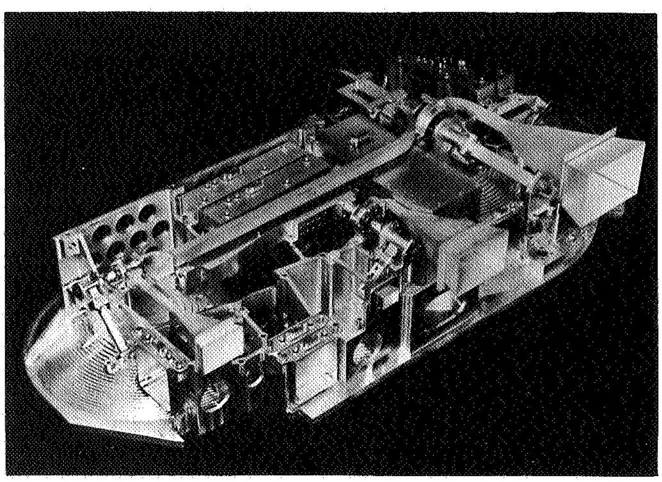


Fig. 16. Back of radiometer assembly prior to thermal control surface treatment

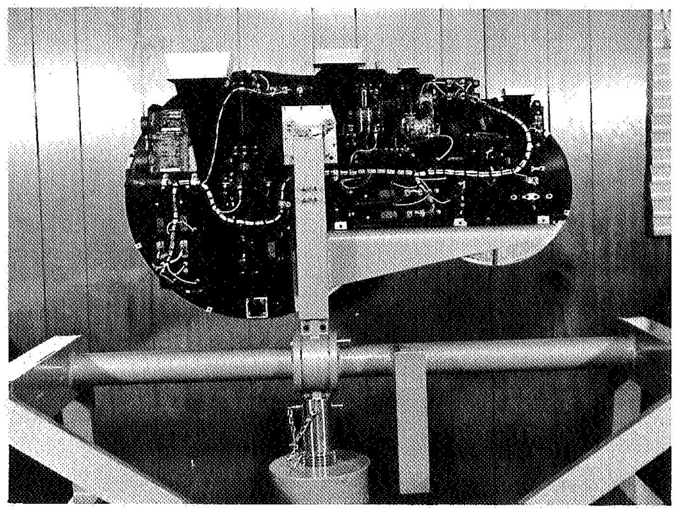


Fig. 17. Back of radiometer chassis assembly following surface treatment

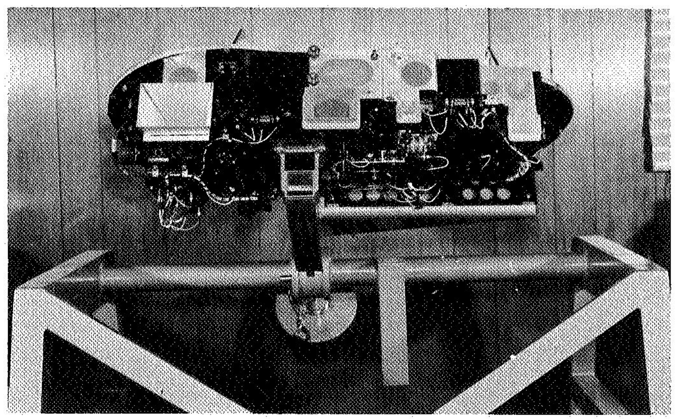


Fig. 18. Back of radiometer assembly ready for installation of thermal control cover

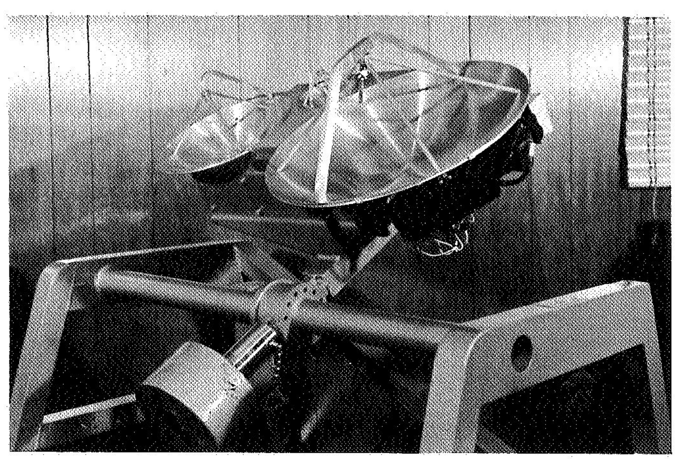


Fig. 19. Front of radiometer assembly showing wave-guide feeds and stepped dishes

The chassis started as a one-piece block of aluminum jig plate 36 × 20 × 6 in., weighing about 400 lb. After machining, the weight of the chassis is 12 lb. The dish faces are stepped parabolas with increments of 1/16 of a wavelength. This was necessary for temperature-control purposes. For the wavelengths of interest the dishes appear as focusing parabolas. For all others the dishes appear as flat reflecting surfaces. This device was to have looked at the Sun for up to 72 hr. Had we not stepped the dishes, a new fabrication technique at the time, the wave-guide focus-feed signal collectors would have melted within about 30 sec. The one-piece construction provides more uniform temperature distribution, closer alignment capabilities, and much less thermal and fabrication distortion.

The noise generator (Fig. 20) is machined in two pieces. Each must match the other within 0.005 in. The mating surfaces are optically flat. You will notice groups of slots. These vary in width from 0.006 to 0.200 in. and were end-milled. This one unit costs $6,000.

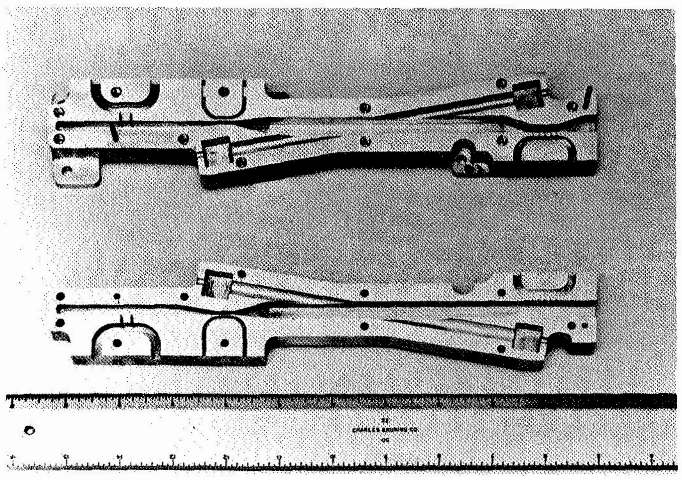


Fig. 20. Radiometer noise-generator coupler

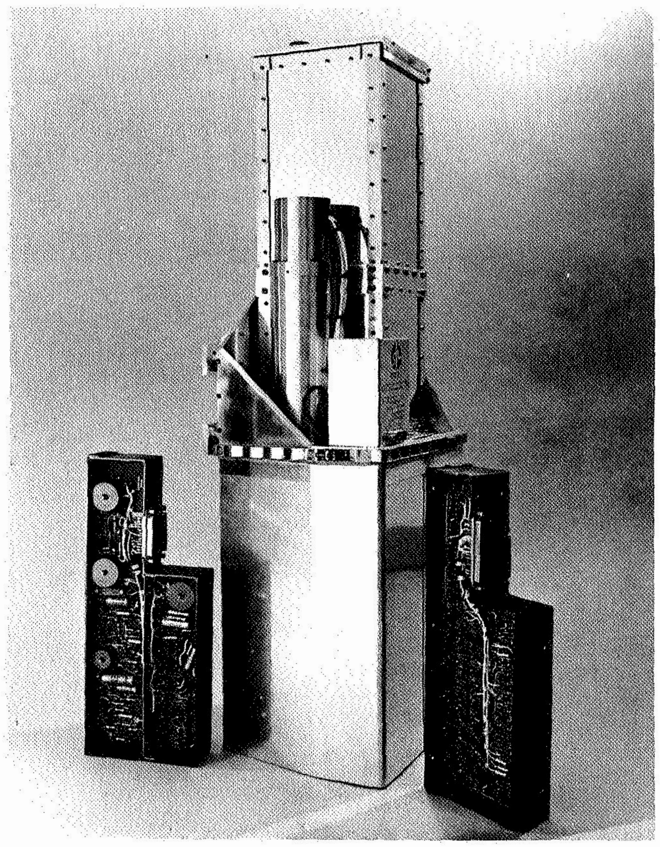


Fig. 21. Ultraviolet spectrometer assembly

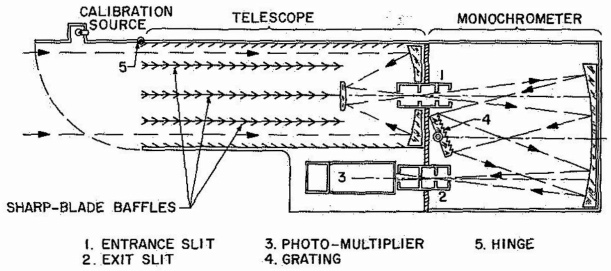


Fig. 22. Ultraviolet spectrometer optical arrangement

C. Ultraviolet Spectrometer

Figures 21, 22, 23, 24, 25, and 26 show various aspects of an ultraviolet spectrometer for a *Mariner* spacecraft. This device will survey the Cytherean atmosphere over wavelengths from 1100 to 5500 Å to determine atmospheric constituents. The device is in three basic parts: an f4 Cassegrainian telescope, an Ebert monochrometer, and supporting electronics logic and control.

Of primary concern in the design were:

1. Protection of the optics during transit to the planet, hence the cover
2. Focus of the optics, elimination of thermal distortion by suspended, compensated design
3. Light tightness of the design
4. Elimination of stray light by baffling
5. Proper mounting, driving, and readout from the diffraction grating
6. Thermal control of the reading photomultiplier tube

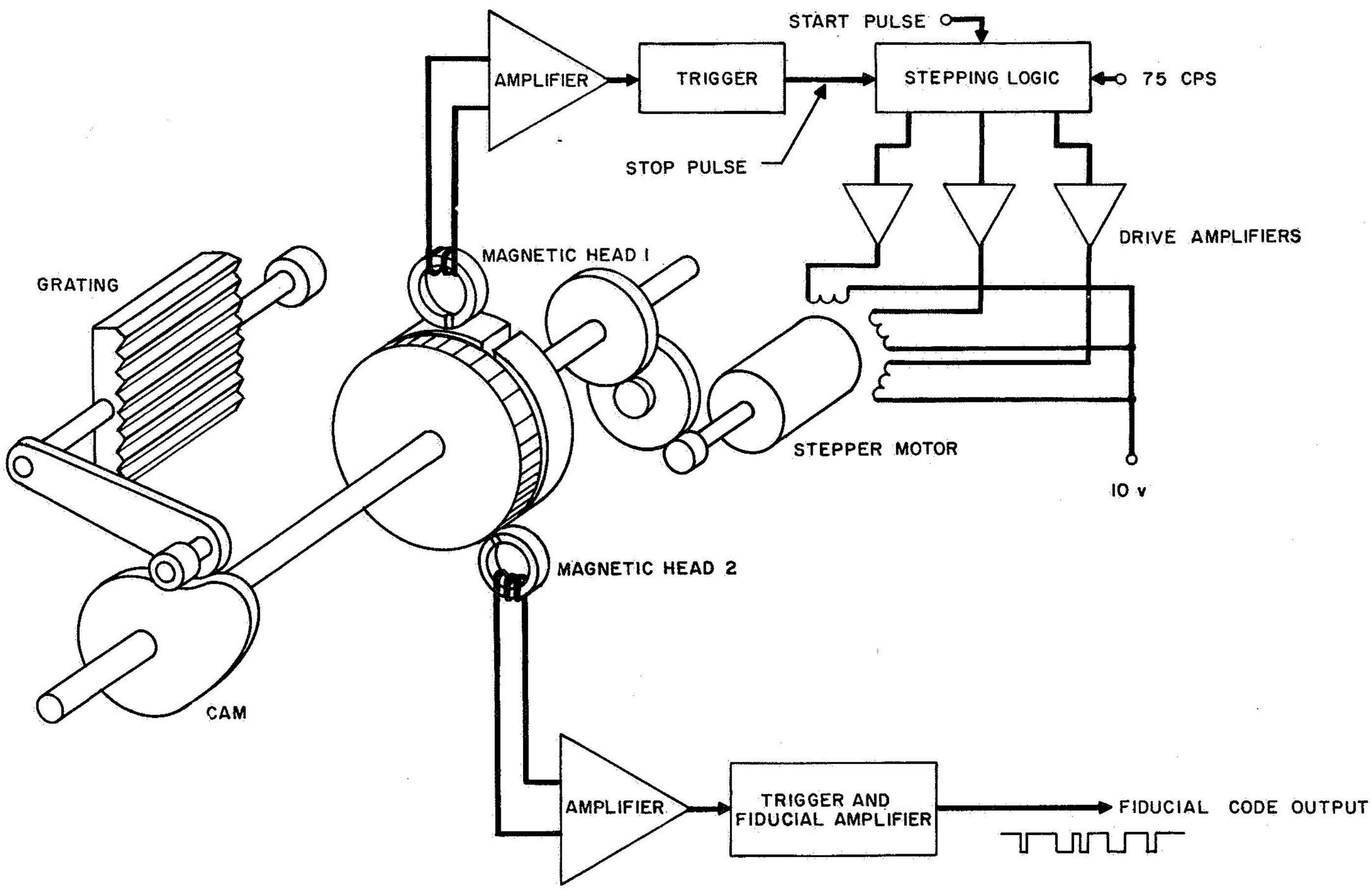


Fig. 23. Ultraviolet grating drive and coding system

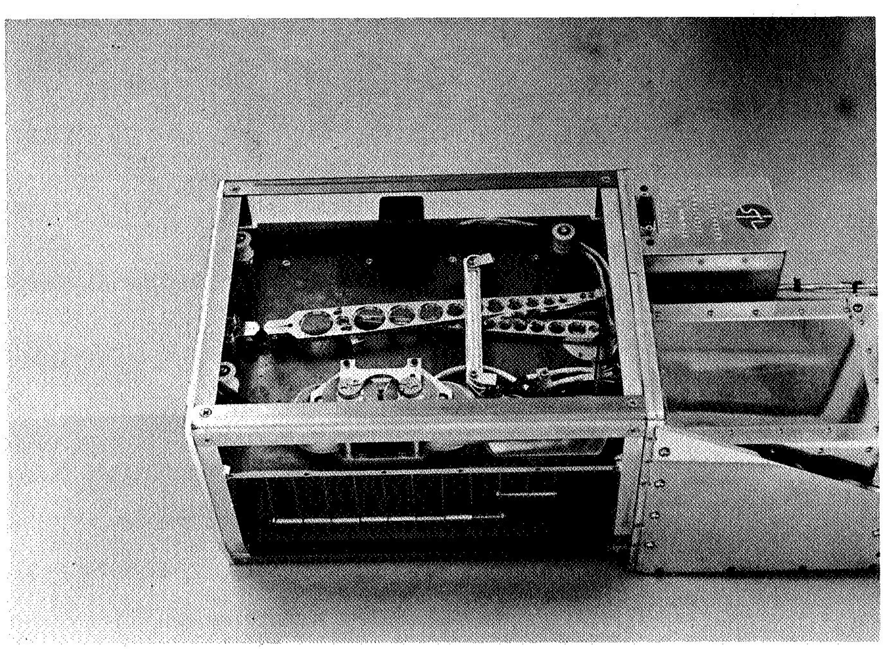


Fig. 24. Ultraviolet spectrometer internal view showing fiducial tape system

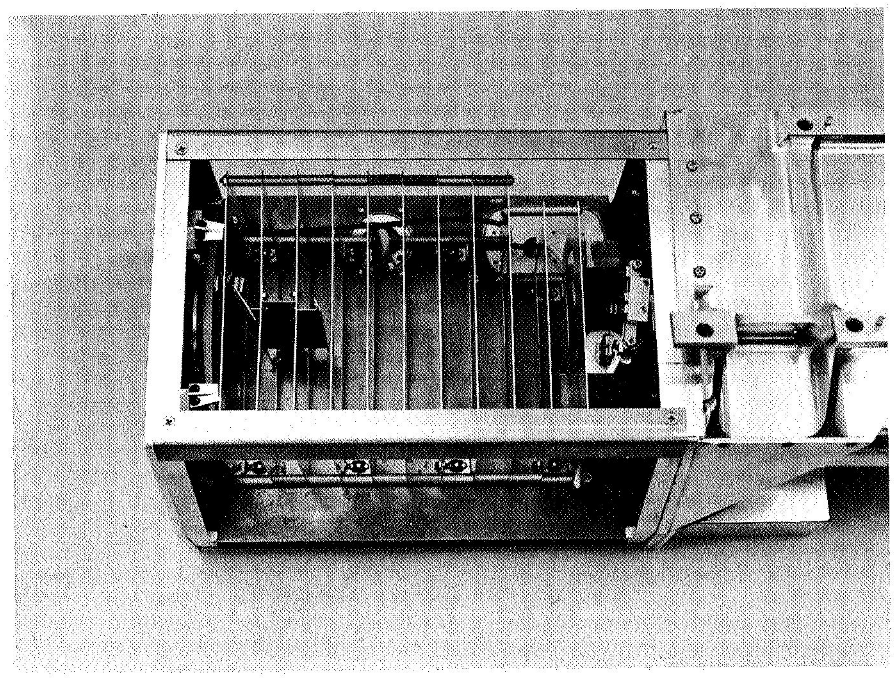


Fig. 25. Ultraviolet spectrometer internal view showing monochrometer optical baffling, diffraction grating, and Ebert mirror (note the massive attach bracket)

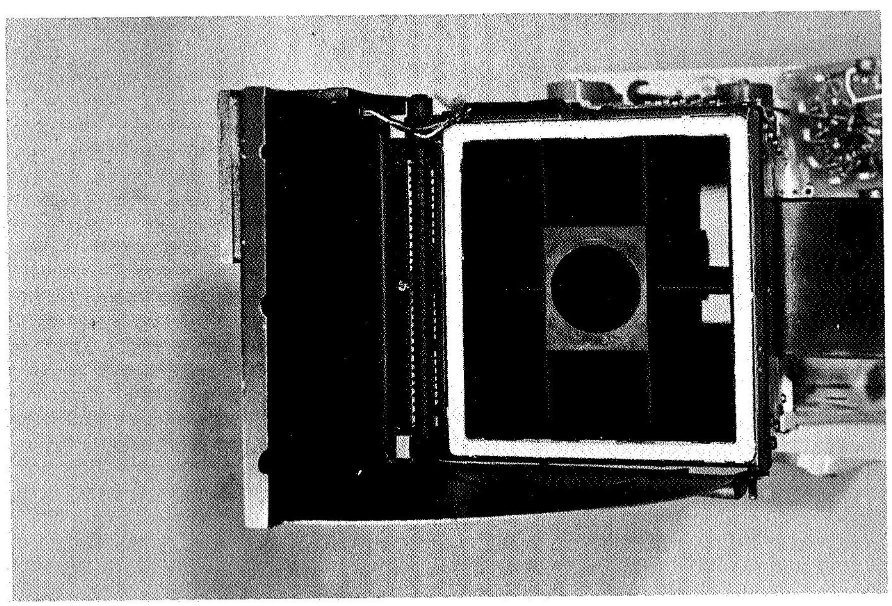


Fig. 26. End view of spectrometer Cassegrainian telescope showing baffling and cover open

IV. CONCLUSION

You can see that the modern engineering problem in the space industry is one of extreme complexity. The industry is at the forefront of technology. It has been my intention with this short discussion to introduce you to some of the details of this highly complex, stimulating area of engineering effort.

Spacecraft Mechanical Subsystems and Mechanisms

RICHARD A. BARLOW

Supervisor, Mechanical Subsystems Group
Spacecraft Development Section

The Jet Propulsion Laboratory is so organized that there are actually two major organizations that become involved in mechanism-type work. One organization is the Guidance and Control Division. Their responsibility covers mechanisms and actuators, which are part of a closed-loop servo system. In the Engineering Mechanics Division, of which we are a part, the work involves only open-loop type actuators and the structures necessary to aim or position such components as science instruments, communications equipment, and power equipment at various attitudes with respect to the spacecraft.

Some examples, typical of our type of design responsibility, can be seen on a model of a *Mariner* spacecraft (see frontispiece). One form of hardware is the folded structure. It is a rigid form such as a mast or panel that has the requirement of being in one position during boost, and then moving to another position and locking later in the flight. Examples of this type of structure are the solar panels and the plasma boom (Fig. 1). Another type of device is the planetary horizontal platform (Fig. 2). These, in general, are multiple-degree of freedom rigid booms for the purpose of tracking a target with two degrees of freedom while providing other degrees of freedom for special instrument scan patterns.

A third type of mechanical subsystem is the extending boom (Fig. 3). The magnetometer boom, developed for the purpose of positioning scientific instruments at long distances from the spacecraft and holding them there in a preknown orientation, is an example of this type of mechanism.

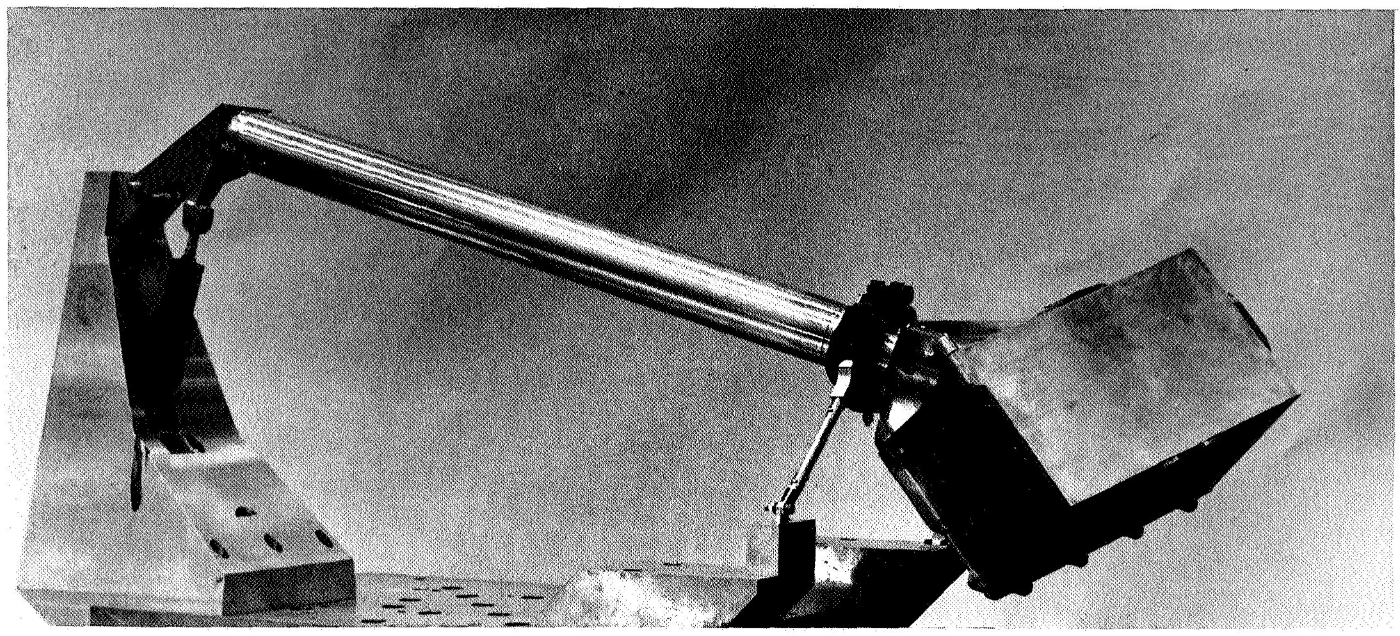


Fig. 1. Plasma boom

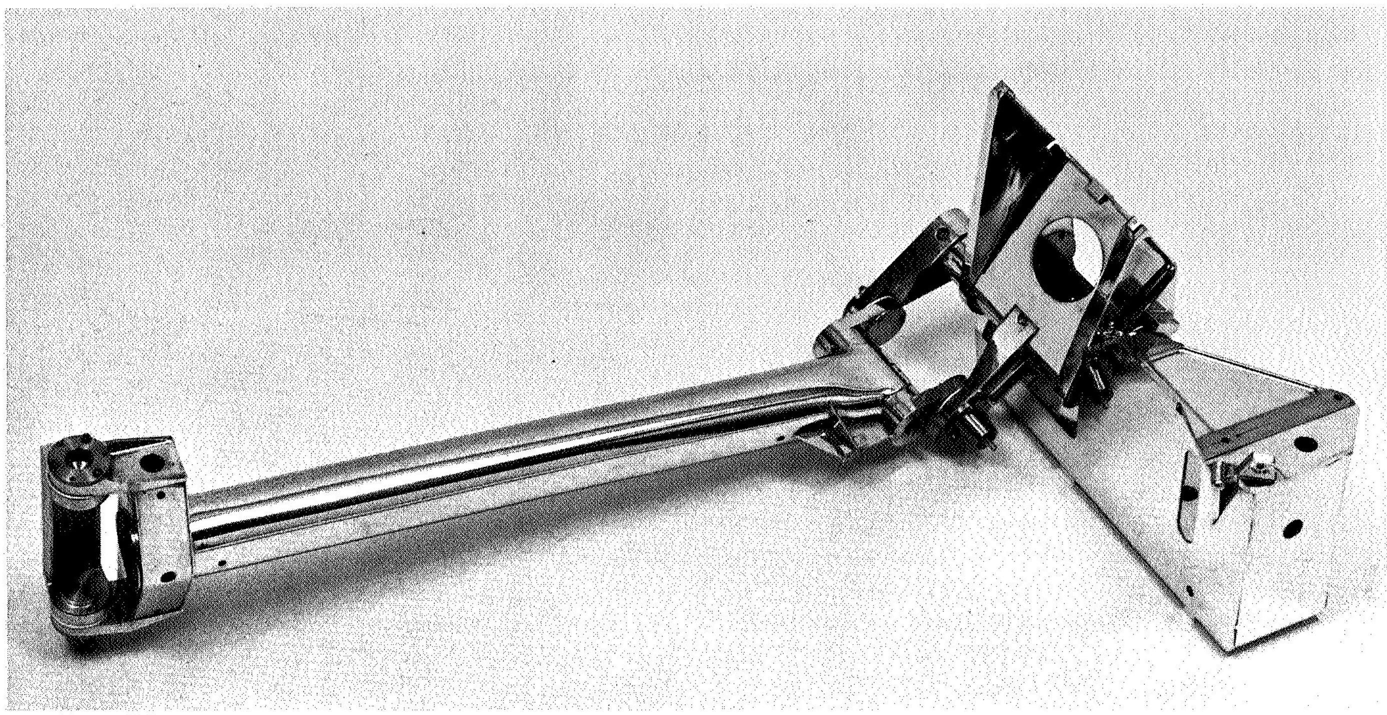


Fig. 2. Planetary boom

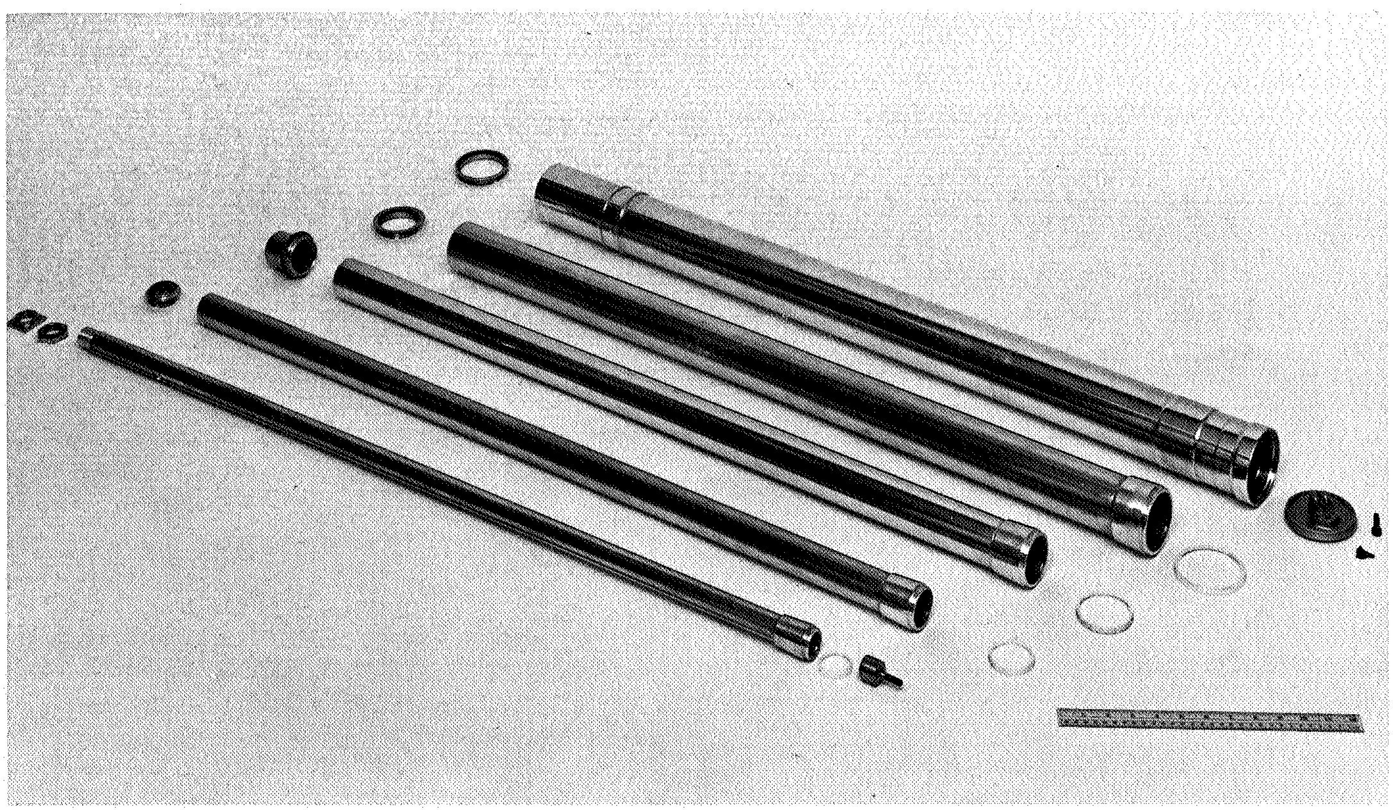


Fig. 3. Magnetometer boom

Lastly, we become involved in the design and development of actuators, both rotary and linear. The radiometer scan actuator is an example of a two-speed rotary actuator developed for making an instrument scan of the target planet as the spacecraft goes by. An example of a linear actuator is the one developed for opening and locking the solar panels and plasma boom. It is a spring-powered actuator that uses self-contained hydraulic oil and orifices for rate control. The importance of rate control in moving any mass on the spacecraft cannot be overemphasized and will be discussed later.

All of the hardware shown is quite common in principle with that used for years in various fields, primarily in aircraft. We are involved in adapting these basic units to a new environment: that of space flight. A brief discussion of the outer-space environment as it affects mechanisms is essential to understanding the reasons why these mechanisms look and operate the way they do.

The first and most commonly known characteristic of the space environment is vacuum. The pressure in interplanetary space is in the order of 10^{-14} mm Hg. This creates problems in providing adequate lubrication for sliding surfaces. All petroleum-type oils and greases have vapor pressures around 10^{-3} mm Hg at room temperatures. This means that the petroleum products will evaporate quite quickly from mechanisms in space, leaving unlubricated surfaces. Since airplane components have been required to operate over extremes in temperature, the aircraft industry over the past years has had to develop a new line of lubricants for this environment. The lubricants which resulted are not petroleums, but are generally classed as synthetics and include the silicones, the diesters, the silanes, and many others. Because many of these lubricants which were developed for high-temperature use have very low vapor pressure they are somewhat useful for spacecraft applications. However, it is important to recognize that each class of the synthetics has some distinct disadvantages when compared to petroleum oils. For example, while the silicone oils have a vapor pressure of about 10^{-7} mm Hg, their lubricating qualities are only fair in comparison to petroleum. Other synthetics have various other problems such as instability, reaction with certain materials, toxicity, undesirable residues, etc. Evaluation of the trade-offs are required for each specific case.

It might be wise at this point to spend just a few moments discussing the function of a lubricant in general. Lubrication provides a low-shear contaminating film between two surfaces that are to move in relation to each other. The theory of friction is based on the development of strong adhesive forces created by the fusion of the contacting asperities of two surfaces that are pressed together. When a load is applied to a bearing, the asperities of the two materials contact and each asperity yields until enough of them are in contact to support the load. The local stress at each junction is equal to the yield stress of the material. When plastic flow occurs at the asperities, the molten materials will weld together if they are soluble in each other and if they are not separated by a contaminating film. When relative motion begins to occur, something must shear at the weld and it is always desirable that the weld shear occur at the original junction rather than within either of the bearing materials. The latter situation results in what we call "galling," and failure of the mechanism follows very quickly. A lubricant assures that the weld, if it was able to occur at all, is quite weak and thus will shear.

In normal atmosphere and in slow-moving mechanisms, oxides of the bearing materials are formed, which can serve as a partial lubricant. For instance, some of the oxides of iron (FeO and Fe_3O_4) are actually reasonably good lubricants as long as they can be maintained on the surface. In normal atmosphere these oxides form in milliseconds, but at a pressure of 10^{-6} mm Hg, they form in seconds, and below 10^{-9} mm Hg, it takes hours or days for the formation of the oxide. Thus the provision of a lubricant or the use of a design using an insoluble pair of bearing materials is absolutely essential for space mechanisms.

There has been considerable discussion of the effects of space travel on mechanical components. Unfortunately, the amount of discussion has far exceeded the actual test work. Therefore, when a piece of flyable hardware must be developed, we are compelled to take the fundamentals as we understand them and make decisions concerning the best way to solve the problems. For instance, to solve the lubrication problem in mechanisms for short-term flights, we have adopted the procedure of using silicone oils and greases on bearing surfaces and then sealing the units to the best of our ability so that the evaporation rate will be low enough to provide lubrication for the length of the flight. We also make extensive use of more or less conventional dry-lubrication techniques such as molybdenum disulphide and Teflon coatings. Many of the softer metals, such as gold, also seem to be promising lubricants. It is very fortunate that Teflon was fully developed before space flight began and that Teflon is not affected by any of the peculiar environments of space. We use Teflon for both its low coefficient of friction and its insulating

ability. Teflon in its natural state has some problems such as cold flow and poor heat conduction, which limit its application. Now, however, there are a considerable number of commercial bushings and bearings on the market that have been compounded to eliminate or alleviate those undesirable characteristics.

Another problem of the space environments in addition to those due to vacuum is that of temperature control. As you realize, all the heat transfer on a spacecraft is by radiation and conduction. The electric motors that we use are conventional BuOrd motors mounted to a bulkhead cantilevered from the output (gearhead) end. Thus, the heat that is developed in the armature, rather than being carried away by air flow, must be both radiated out through the motor's field and housing and conducted out through the bearings, down through the gearhead housing and into the mount. To assist the radiation, the proper coatings are applied to the motor and to the inside and outside of the actuator housing. In general, mechanisms are designed to run quite cold in order to inhibit the evaporation rate of the oils.

A third aspect of the space environment is radiation. Ultraviolet adversely affects some of the materials which we might otherwise like to use. Fortunately, as stated previously, Teflon is not affected by radiation and we can use it without reservation in this respect.

A very serious consideration in the design of mechanical systems for space travel is the "inconvenience" of the laws of physics. Reactions, momentums and inertia, and the absence of gravity are critical considerations in mechanisms design. For example, the latter is important whenever we consider circulating-fluid lubricant systems. Obviously, under a zero-g environment, control of the fluid is a problem and conventional sump operation is impossible. Inertia and reactions become a problem when we accelerate and/or decelerate components such as magnetometer booms and boom-mounted instrument packages, which have an Earth weight in the order of 80 to 100 lb. The situation can become that of the tail wagging the dog if careful control of acceleration and deceleration rates is not maintained. Forces can never be put into the spacecraft which either the attitude control system cannot correct at all or which necessitate the use of a large amount of attitude-control gas.

Designing equipment that has to be efficient for space flight under zero-g conditions of no loads other than inertia and yet be able to survive the vibration and thrust during the boost phase becomes a situation of balancing conflicting requirements. Most of the mechanical or mechanisms equipment is mounted on the spacecraft structure so that it sees a vibration input as high as 7 to 8 times the 15-g input from the booster. This vibration can cause bearing brinnelling, fatigue damage in structures, galling of contacting gear teeth or other touching metals, and impact damage to parts that have out-of-phase deflections greater than the distance between them.

Although it is not part of the space mission as such, the preflight testing can be a more severe service than the flight itself. We have had several cases of units designed for space flight that would work excellently in space, but are not able to survive the handling and hours of operation during the check-out procedure.

As an example of the process that is involved in the development of hardware and some of the problems that arise, we will trace the history of the development of the magnetometer boom. The design started off in a nebulous fashion, with much negotiating and bargaining to develop the design criteria. For purposes of demonstration, the design requirements are listed in Fig. 4. The manner of approach to such a problem is outlined in Fig. 5. Step 1 can be considered as covered in the statement of the problem. Step 2 of the design process, listing constraints and their importance, is presented in Fig. 6.

The problem and the constraints were analyzed and it was concluded that the only configuration that could meet most of the constraints and the extension ratio required to get within the shroud and also extend to over 20 ft would be a multiple-stage extending boom.

The folded and extended lengths are a function of the number of stages. The number of stages allowable is limited by the stiffness of the smallest-diameter stage on one end and the allowable (spacewise) diameter of the base tube on the other. Obviously, both the weight and frequency constraints also must be considered at this point in the design.

A six-stage boom slightly over 4 ft long closed and 26 ft long extended was selected. The outside diameter of the base tube was 3¼ in. and the inner tube was 1$\frac{1}{16}$ in. With the basic configuration roughed out, the detail design was started.

First of all, the energy source for extension and the latching technique for both the closed and the extended

DESIGN A DEVICE WHICH WILL SATISFY THE FOLLOWING REQUIREMENTS:

1. POSITION A 2-lb INSTRUMENT PACKAGE (CONFIGURATION NOT FIRM) AS FAR AS POSSIBLE (AT LEAST 20 FEET) FROM THE SPACECRAFT, WITH ANOTHER SMALLER PACKAGE SEVERAL FEET FROM THE FIRST.
2. THE POSITION OF THE INSTRUMENTS MUST BE WITHIN ±2° OF A KNOWN ORIENTATION.
3. THE SMALLER INSTRUMENT IS A MAGNETOMETER, THUS NO MAGNETIC MATERIALS CAN BE USED IN THE BOOM.
4. ABOUT TEN No. 22 SHIELDED WIRES MUST BE CONNECTED BETWEEN THE INSTRUMENTS AND THE SPACECRAFT.
5. WEIGHT RESTRICTION: 10 lb EXCLUDING INSTRUMENTS AND CABLING.
6. RESONANT FREQUENCY MUST BE ABOVE 1 1/2 cps EXTENDED.
7. ENVELOPE UNDEFINED.
8. NO MORE THAN ONE COMMAND SIGNAL CAN BE USED FOR EXTENSION.
9. CONSUME MINIMUM ELECTRICAL POWER.

Fig. 4. The problem

1. DETERMINE THE GENERAL DESIGN AND INTERFACE CONSTRAINTS
2. DETERMINE THE RELATIVE IMPORTANCE OF THE CONSTRAINTS
3. ROUGH OUT THE CONCEPT TO BE USED AND THE APPROACH TO THE PROBLEM
4. START PRELIMINARY LAYOUTS
5. PERFORM NECESSARY CALCULATIONS
6. CONTINUOUS REFINEMENT OF THE DESIGN AND RENEGOTIATION OF INTERFACES AND CONSTRAINTS
7. BUILD PROTOTYPE
8. EVALUATE
9. REDESIGN AS REQUIRED

Fig. 5. The design process

positions had to be determined. Cold gas (N_2) was selected for the energy source and pyrotechnics for the initiator. The decision to use pyrotechnics was based on the electrical power restrictions and the fact that a valve and N_2 tank which would do the job had already been developed for the midcourse motor. Incidentally, the task of designing, developing, and qualifying a valve would have been a major project in itself. Latching in the closed position by a technique not requiring a separate command and additional electrical energy turned out to be quite a challenge.

Briefly, the latch to keep the boom closed is a rod between the end of the smallest stage and the base plate. The rod has a notch near the base into which a sear is fitted, holding the rod to the base. Tensioning is done from the exterior at the head end of the cylinder. Upon firing the squib, gas is routed to a piston that releases the latch and then uncovers the port through the rate-control metering orifice into the boom. The extended-position latch consists of locking tapers backed up by grooves at the end of the stroke into which the seal lip drops, preventing bounce-back if the tapers fail to lock.

The most severe restrictions on the boom were those imposed by the attitude-control system. They included the c.g. shift, change in moment of inertia, reaction rates imposed during extension, and the resonant frequency after full extension. Leakage of gas after extension also had to be controlled because such leaks imparted thrust to the spacecraft. Actually, these restrictions presented fairly straightforward dynamics problems except that the capabilities of the attitude-control system were not really known at the time.

About the time the design was complete, the attitude-control engineers blew the whistle. Their analysis of the attitude-control system indicated that the resonant frequency of the extended boom had to stay out of the ½ to 2½-cps range and that good damping would be required. About the same time it was determined that the spacecraft c.g. was out of tolerance and the boom had to be heavier for counterbalance. As a result, design more or less had to be started over. Two stages were eliminated from the boom and the wall thickness of the remaining sections was increased from 0.025 to 0.060 in. As a result, the boom was shortened to a total extended length of 17.4 ft and the weight was increased to 10 lb. The resulting resonant frequency was 2.2 cps, which was decided to be adequate. (Figure 3 illustrates a magnetometer boom.)

The special equipment for the qualification tests and the flight-acceptance tests turned out to be almost as

1. ATTITUDE CONTROL
   - A. CG SHIFT UPON EXTENSION
   - B. MOMENT OF INERTIA CHANGE UPON EXTENSION
   - C. ROLL, PITCH AND YAW RATES INTO SPACECRAFT UPON EXTENSION
   - D. EXTENDED RESONANT FREQUENCY AND DAMPING CHARACTERISTICS
2. ENVIRONMENT
   - A. VACUUM
   - B. TEMPERATURE CONTROL
   - C. BOOST VIBRATION
3. ENVELOPE
   - A. FIT ON SPACECRAFT, UNDER SHROUD
   - B. EXTENSION RATIO
4. CABLE FEED-OUT SYSTEM
5. PREFLIGHT EVALUATION AND CHECKOUT

Fig. 6. Constraints

much of a design problem as the boom. Since the final pressure in the boom was about 2 psia, the boom was obviously incapable of full extension in atmosphere. Also, the boom could not be extended in a 1-g field for rate calibration because the condition caused by the weight of the boom resulted in unflightlike friction. It was necessary to develop a 24-ft-long vacuum chamber with an overhead rail system to allow normal firing of the boom and to support it as it extended. Figures 7 and 8 show this type of vacuum chamber.

The purpose of this brief summary of the history of one assembly is to demonstrate that hardware design is a long series of design, negotiate, design, negotiate, etc. The problem is that everyone starts together, and the people who are working on the hardware that has interfaces with yours usually are no further along than you, and are in no position to give all the important answers at the time they are needed.

Once the design phase is over and the drawings have been released, one of the engineer's greatest trial periods is yet to come—that of getting the parts made, evaluated, and delivered. During the design it is necessary to do a complete tolerance analysis of the assembly to be sure that it can be put together within itself, and then that it will fit in proper alignment with the mating parts. This requires judgment, so that tolerances are set up that are realistic from a manufacturing standpoint and yet can fulfill the engineering requirements. Unfortunately, parts are not always made within the tolerances specified on the drawings. The engineer is then faced with a decision that involves judgment and experience: that of buying or rejecting the part.

Several factors enter into making this decision, a major one being the pressure for delivering the hardware on schedule. In most cases, to reject a part means a slip in the delivery schedule. On the other hand, there is the inherent desire for perfect parts. An engineer never likes to buy substandard components. Assuming that schedule does not allow an immediate "reject" decision, a careful analysis of the consequences of the deviation must be conducted. A deviation could result in any combination of stress problems, alignment problems, power requirement problems, degraded life, decreased reliability, etc. Another factor that must be considered is the probability of obtaining good parts on a second try. It is sometimes possible that the parts on hand represent a best effort, and the probability of getting in-tolerance parts on a second try is no better than it was on a first try.

After the device or subsystem is built up and assembled, whether it is completely according to specification or not, it must be submitted to a long series of tests. These tests fall into two categories: the Test Approval or design evaluation test, and the Flight Approval or specific part performance test. In most cases, the engineer who is responsible for the hardware is also responsible for setting up the test program within certain guide lines. This responsibility requires judgment so that the test levels are neither so severe as to wreck the assembly, nor so mild as not to be a valid test.

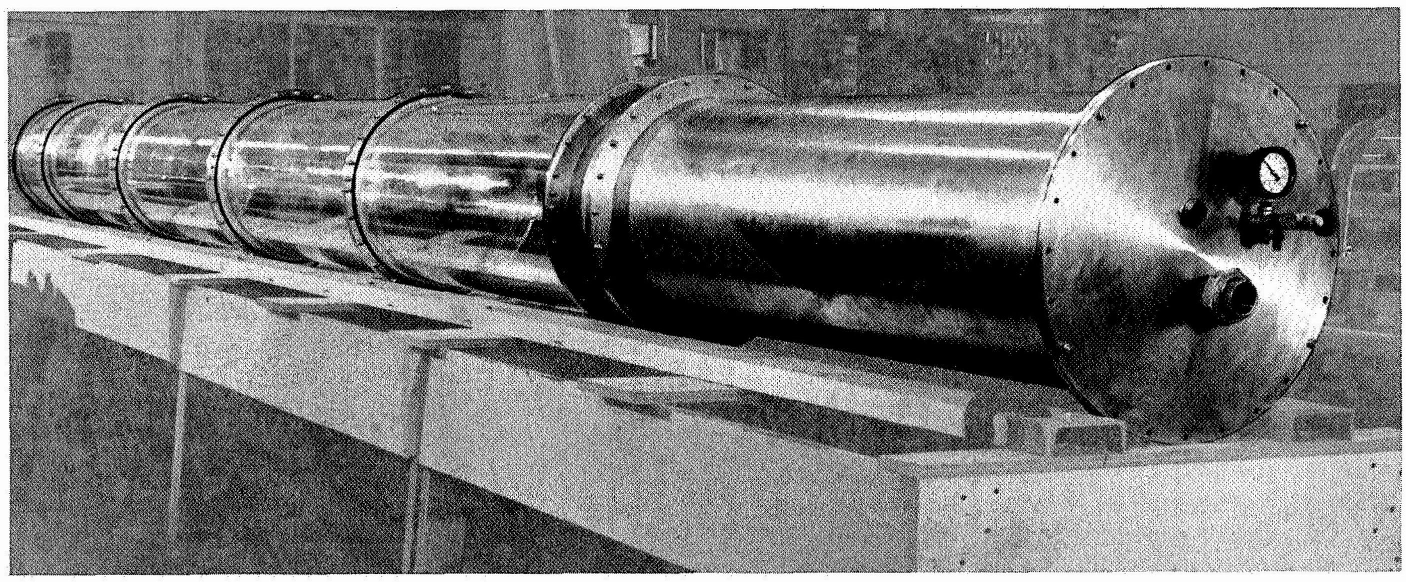


Fig. 7. Vacuum chamber

Many assemblies fall in the "one-shot" device category, thus creating a special evaluation problem. In these cases we normally use statistics for predicting reliability on a sampling basis.

In conclusion, it should be pointed out that there is no real substitute for technical competence, but experience and judgment are of almost equal necessity in the overall job. It should be recognized that there is seldom only one way to accomplish a design task. Many non-technical factors enter into the solution of a design problem. In the spacecraft business, schedule is always extremely important and dictates that the first design must be right; therefore, in spite of a desire to create a new ingenious device, it is mandatory that a path be chosen to provide the highest level of confidence of immediate success. In many situations the problem of testing or preflight evaluation is a big element in the design of the hardware. Surmounting those factors are the emotions and the prejudices of the engineer himself and those he works for and around.

In the design process there is only one thing that can always be depended upon: no matter what method you have chosen to accomplish the job and how much ingenuity and effort have been applied, once the hardware exists, everyone will ask why you did it in the manner you did, rather than in some other.

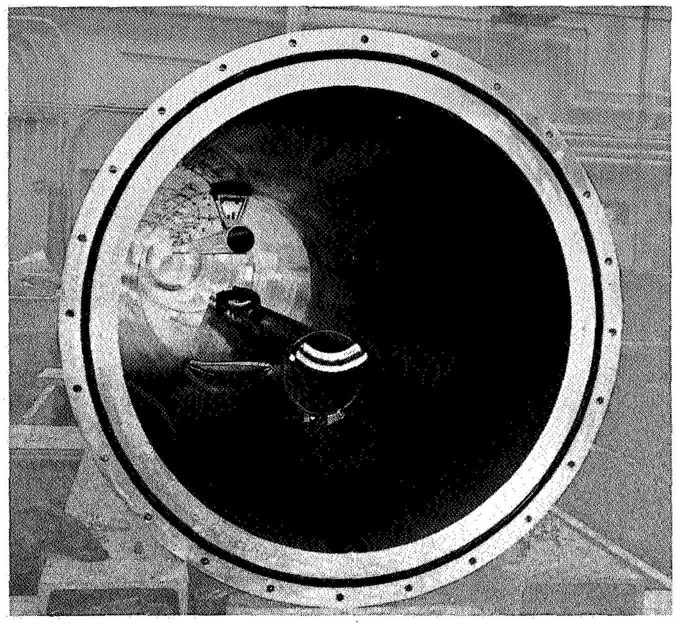


Fig. 8. Interior detail of vacuum chamber

N67 11286

The Application of Structural Engineering to Spacecraft Design

NEIL E. JOHNSON

Senior Development Engineer, Structures Group
Applied Mechanics Section

This lecture concerns the application of structural engineering to spacecraft design. During the course of this discussion, we will be talking, in rather general terms, about some of the processes that we go through in actually performing the structural design, analysis, and testing of a spacecraft. Particularly, we will be talking about the *Mariner 1* spacecraft which you have been introduced to in some of the earlier seminars in the series.

Today we are going to talk about the major structural components of the spacecraft (see frontispiece), including the primary hexagonal structure, which is called the bus; the solar panels, of which I have a sample that you can look at after the talk; the superstructure; and the high-gain antenna. Some of the components, such as the radiometer and the omnidirectional antenna on the top, may be classed as secondary structural components, and for this reason will not be discussed here.

Before delving into some of the more interesting problems of the structural design, I have outlined a series of steps that is intended to symbolize the processes, or the phases of design, that we go through (see Fig. 1).

I. PRELIMINARY DESIGN
II. INITIAL STRUCTURAL DESIGN
III. DEVELOPMENT TESTS OF PROTOTYPE STRUCTURES
IV. FINAL STRUCTURAL DESIGN
V. STRUCTURAL TESTS OF COMPLETE SPACECRAFT
VI. REDESIGN IF NECESSARY (OR OPTIMIZATION)

Fig. 1. Phases of structural design and design verification

The preliminary design phase has been reasonably well covered in some of the earlier seminars, particularly the one on "Spacecraft Design." It is during this period of the design that the configuration of the spacecraft and the environment through which the spacecraft must perform, and the corresponding loads acting on the structure, are established.

These loads might include:

a. Static thrust of the booster

b. Vibration from the booster

c. Aerodynamic noise

d. Thermal effects of aerodynamic heating and cryogenic propellants

e. Maneuver of the spacecraft or booster

You will observe as we progress through this discussion that it appears as if a majority of the analyses and tests performed on *Mariner 1* were related to the behavior of the spacecraft in a vibration environment. It is not intended that the effects of other environments be neglected; however, for reasons brought out in part as we develop the design process, the vibration environment was of primary interest in this case.

The step following preliminary design is referred to as the initial structural design. Once the configuration has been established, we actually detail the size of the members and various components in the structure.

Rather than taking the whole spacecraft as one very complex structural analysis problem, it is often more efficient to separate the problem into definable units. For example, the solar panels can be handled as a separate detailed problem. The superstructure can be analyzed individually. We try to uncouple the characteristics of these various parts of the structure so that we can look at

them on a singular basis, especially during the initial structural design phase. As stated before, most of the *Mariner 1* analyses were performed on the spacecraft in its vibration environment. This was due mostly to the fact that we took an earlier spacecraft, the *Ranger*, and modified it. We were familiar with what the *Ranger* had done while in a static load environment, for example, so we felt that we could perhaps eliminate some phases of the static analysis and testing.

Once the initial design and analysis has been performed, many of the major components are fabricated and tested to the environments found on the spacecraft.

After performing these developmental tests, we get into the final structural design. During this phase we apply all of our experience and insert a certain degree of sophistication and optimization into the design. We then build our prototype structure from this design. The next step is to run a series of tests on the actual flight prototype structure. Following this series of tests during the *Mariner 1* program, we redesigned those elements that suffered failure during the qualification tests and used those we had left as originally designed. This was particularly true on this specific program for we were working with a very tight schedule. On programs with more time available, this period could be used to further optimize the design.

Next, we go into more detail in those design steps following preliminary design. Figure 2 shows our method of setting up the superstructure idealization on *Mariner 1* in order to perform an analysis. This is a fairly straightforward design, and from the structural engineer's point of view, a simple problem. The design was developed from the following constraints. It was necessary to have various items supported on the *Mariner 1* at three different levels. Instruments such as the magnetometer and the radiometer had to be supported at certain levels, and the top end of the solar panels had to be supported at the middle level when they were in their stowed position. At the top level, the omnidirectional antenna had to be supported. A space frame was a logical choice of structure to satisfy these conditions.

Once we determined the appearance of the structure, we resolved the sizes of each of the members on a basis of the following criteria. First, each of the members should meet its allowable stress. Second, each of the members may be buckling under axial load once the allowable stress is exceeded. Finally, the natural frequency of the whole superstructure had to be uncoupled from

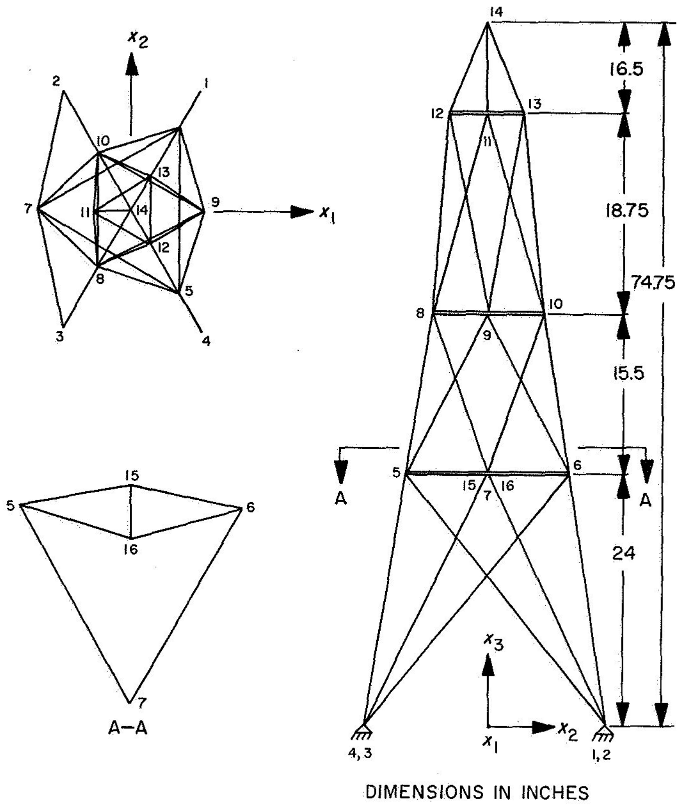


Fig. 2. Idealization of superstructure for structural analysis

the other parts of the structure around it. We particularly wanted the frequency of the solar panels and the frequency of the superstructure to be uncoupled. With these criteria we idealized our frame as a pin-jointed structure. Then, with the aid of the JPL IBM 7090 stiffness matrix analysis computer program,[1] we determined the lowest six natural frequencies of the superstructure, together with their respective mode shapes and the loads due to both the static environment and vibration environment. After several iterations, rather a good design for the structure evolved.

Figure 3 shows the bus structure underneath the superstructure. This structure presented a more complex problem. It is again basically a space frame structure; however, in this case, the effect of joint fixity can be expected to be significant. In the case of the superstructure, with its members relatively long and of small cross-sectional area, the effect of joint fixity might only be on the order of 10 or 20%. We tried to analyze the problem

[1]*Stiffness Matrix Structural Analysis*, by R. R. Batchelder and B. K. Wada, Technical Memorandum No. 33-75, Jet Propulsion Laboratory, Pasadena, February 12, 1962.

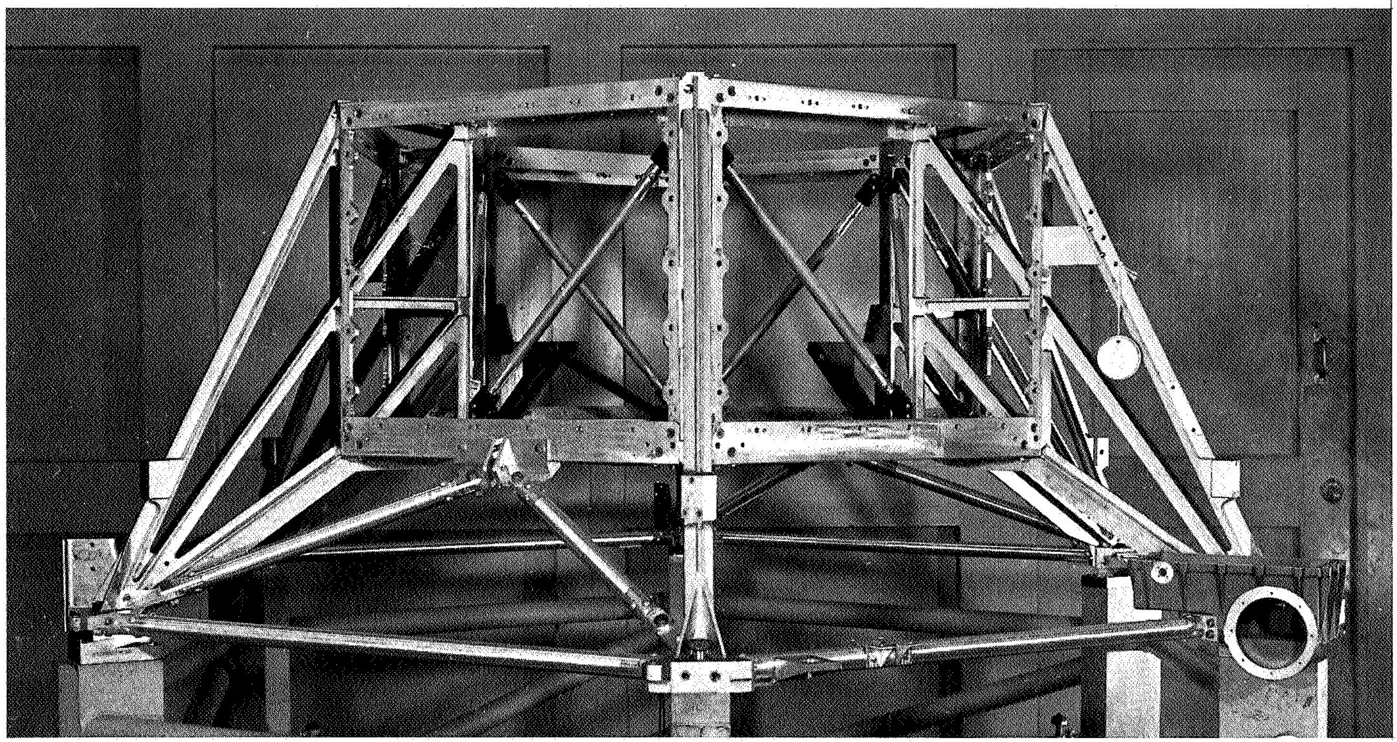


Fig. 3. *Mariner 1* spacecraft bus

in the same way that we analyzed that of the superstructure by again using our computer program; however, we did not expect the results to be as good as those for the larger structure. Basically, this structure is a modification of the *Ranger 3* bus. *Ranger 3* had shear panels in various parts, particularly in the back faces and at the top and bottom of the structure. The shear panels propounded several problems. One of the problems was due to a requirement from the temperature-control people to provide a surface in the back of the electronic chassis that would transmit as much heat as possible into the interior. Using a very practical engineering approach, we put in some diagonal tubular members which also fitted in well with the concept of a space frame which we were trying to analyze. Another major change differentiating *Mariner 1* from *Ranger 3* was to relocate the cable trough in order to permit the installation of the midcourse motor from below. The tie rods at the base of the bus were also redesigned from gold-plated magnesium to polished aluminum for ease in fabrication.

The next structure on which we ran an analysis was the solar panel structure. Basically, the solar panel is a nonisotropic rectangular plate that is supported at six points (Fig. 4). This plate is a corrugated structure with stiffeners running transversely to the corrugation. We did simplify our design, to a certain extent, in that we had an opportunity to uncouple the solar panel frequencies from the rest of the structure, thereby enabling us to analyze it as a separate item. If its frequencies had been coupled with the superstructure, it would have been much more difficult to handle. Moreover, we had a thermal stress problem that came from the *Agena* booster which supported the spacecraft, in that the distance from point No. 29 to No. 15 would change (see Fig. 4). To accommodate this situation, we allowed an expansion gap at the hinge points which permitted the spacecraft to expand without introducing any loads into the solar panel. In order to analyze the structure we again referred to our stiffness matrix computer program. For this problem, we used an option which idealized our plate as a grid with each member idealized as a beam. Using this idealization, we went into the program and came out with loads in each of the beams which could be transposed back to get loads into each of the grid members. As before, we also got the dynamic characteristics of the panel.

I could go on to some of the other components, but the analyses are generally performed in more or less the same manner. However, one comment is in order at this point. While most of the large pieces of structure can be, and generally are, analyzed either with the aid of a computer

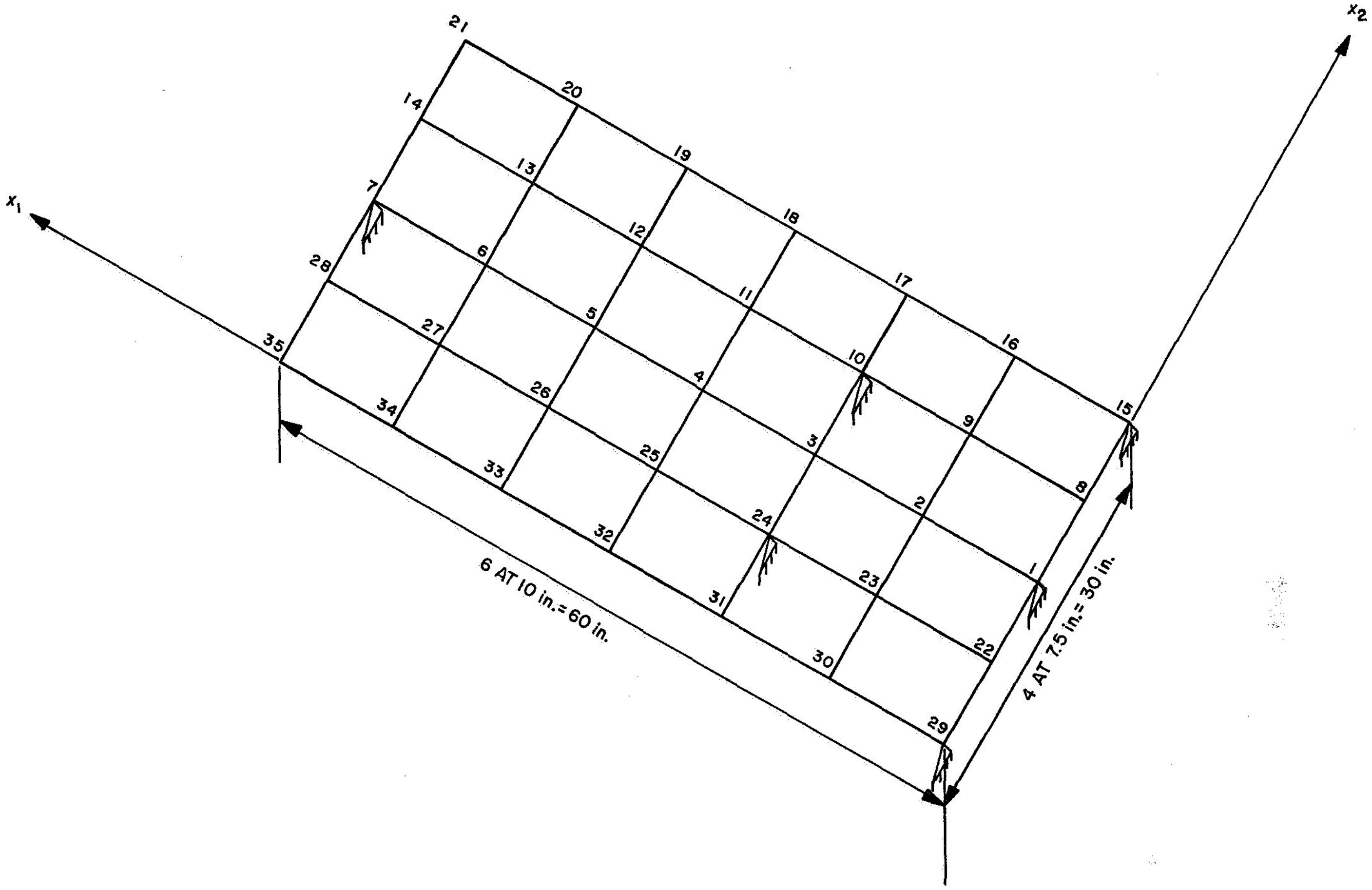


Fig. 4. Idealization of solar panels for structural analysis

program or with complicated hand calculations, a great many structures are analyzed using elementary "back-of-the-envelope" techniques. The ability of an engineer to determine the complexity of analysis required for a specific problem is a valuable asset.

Once we have performed all these analyses, we actually build sample structures and perform tests. Figure 5 shows an example of one of the developmental test setups for prototype structures. The superstructure is mounted on an oil table which is attached to a Ling electromagnetic shaker. It can be seen that we have dummy masses where various experiments were to be mounted. An I-beam, intended to simulate the radiometer, was supported at one point. When we shook this structure in a plane normal to the solar panels, the solar panel load acted on the superstructure, so we placed a dummy mass on the superstructure to simulate the effect of the solar panels. When we shook the structure in the other planes, the solar panel loads did not go into the superstructure, so we removed the mass from the system. We performed tests in each of three orthogonal directions. This structure was to be riveted, and it is on the actual flight spacecraft; however, for schedule reasons, the most convenient thing to do was to weld all the joints for this test. We hoped this would work, but the actual stress in the welded area exceeded the allowable stress of the weld, and during one of the shake tests, we broke a joint. In order to get enough engineering data, we rewelded the structure and continued our test at a reduced load level. The final satisfactory outcome of this test illustrated the fact that even though we did not have exactly what we were going to fly, it was close enough to provide much valuable information for the final design.

Figure 6 illustrates the *Mariner* prototype superstructure development test setup. This is a more sophisticated approach to constructing a test than that performed on the *Mariner 1* superstructure. In that test we had included only a mass to simulate the load of the solar panels. With the test for the prototype, we went one step further. We actually built simulated panels through the

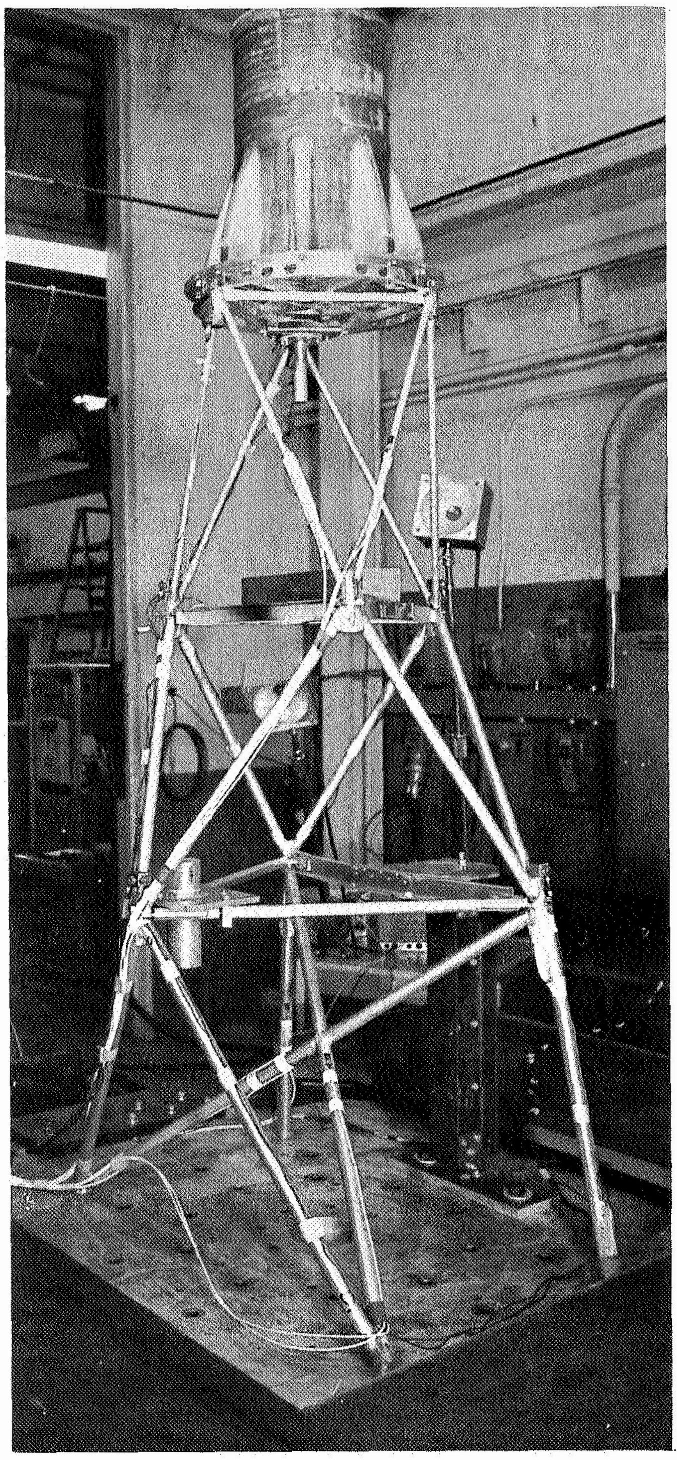


Fig. 5. *Mariner 1* superstructure development test setup

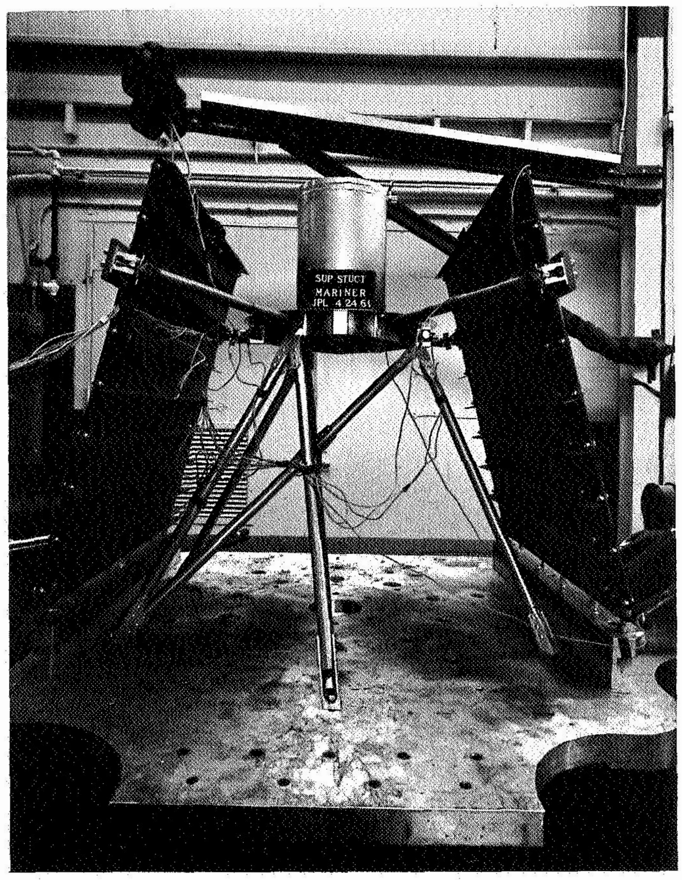


Fig. 6. *Mariner* prototype superstructure development test setup

use of steel plates stiffened in such a way that the first longitudinal bending mode and the first transverse bending mode were correct. We also simulated the mass distribution and attempted a fair approximation of the damping in the system. We had to make a number of assumptions because the panels had not been built at that time. When the test was completed, the consensus was that while our answers were not very accurate, they were close enough, especially since we had applied loads higher than would be anticipated in reality. Therefore, this was a conservative test.

Figure 7 demonstrates a test that we performed on a solar panel using a modal survey technique. This is just one of a series of tests that we ran on this solar panel. We ran a thermal test because the solar panel itself sees a very high temperature when the Sun's heat beats down on the solar cells and the thermal-control people were afraid of problems such as buckling on the panel surfaces. We also ran vibration tests by putting the whole solar panel by itself on the shake table. In addition, we ran a series of modal survey tests taking a series of small 25-lb shakers and applying sinusoidal vibration excitations to the structure at various points. For the case illustrated, we tried to excite a torsion mode on the end of the panel.

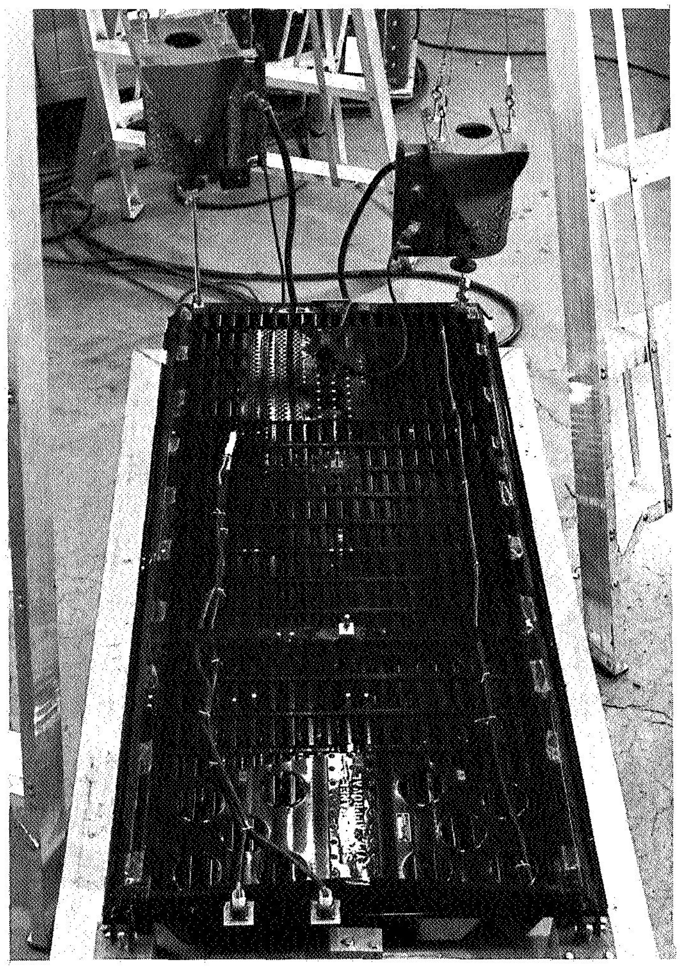


Fig. 7. ***Mariner 1*** **solar panel TA modal survey test setup**

Using two shakers, we swept through the frequency range to define the natural frequency of this system. Once this frequency was discovered, and while the structure was vibrating in its normal mode, we took two accelerometers and moved them around the structure, reading the voltage off through the accelerometers. Since the voltage output of the accelerometers is proportional to displacement, we were able to determine exactly what the mode shape was. Moreover, after this was completed, we removed the armature current to all the shakers simultaneously, and recorded the vibration decay on an oscillograph record. Using the log-decrement concept we then determined the damping in our system. Using all this information, we were able to go back and predict, experimentally, the behavior of our structure.

The high-gain antenna static test (Fig. 8) is interjected mainly to illustrate the fact that all our tests are not vibration tests. It is, however, one of the few static tests that we have performed on this program. It illustrates the type of static-load test we might want to perform. In this case, we had a high-gain antenna, which was the antenna underneath the spacecraft, mounted in a big supporting frame. Using dead weight, we applied our thrust load, which amounted to 10 g, to the antenna feed. This feed weighed about 10 lb. Then also, through a pulley system, we applied a side load of about 2 g to simulate the effects of the booster maneuver load on the spacecraft. Then, in applying these loads, we measured influence coefficients, and finally determined the structural adequacy of the antenna to survive the expected flight loads.

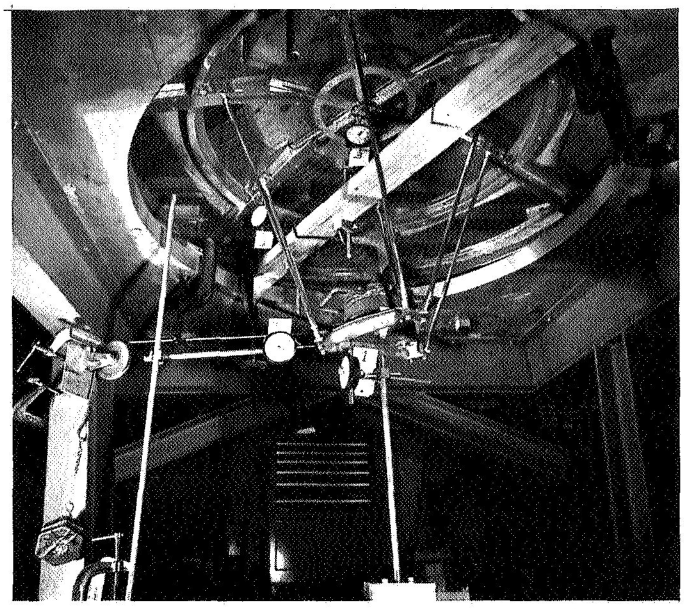


Fig. 8. ***Mariner 1*** **high-gain antenna static test setup**

After we finished the developmental testing phase of our program, we examined all the test data and all the information from our analyses, and using the experience we now had, came up with our final spacecraft design. Also during this final spacecraft design phase, we tried to begin to make an analysis of the entire spacecraft as one complete unit, realizing that this was a complex problem, and that probably we would not hit the answer right on the head. After we had finished this analysis, we then went ahead and built the flight design structure (Fig. 9). This spacecraft is identical, structurally, to the spacecraft that will actually fly to Venus this year. At the time the flight design structure was built, there were some components for which we did not have complete details, so we simulated them with masses as well as we could. For example, we had to put masses on the back of the radiometer at various points to simulate what we thought the radiometer load would be. We inserted masses at various

points on the superstructure to simulate experiments. Each of the main electronic chassis was completely filled with masses to simulate the various electronic components of the spacecraft. We took this particular model, which we call the structural test model, and performed a series of tests on it.

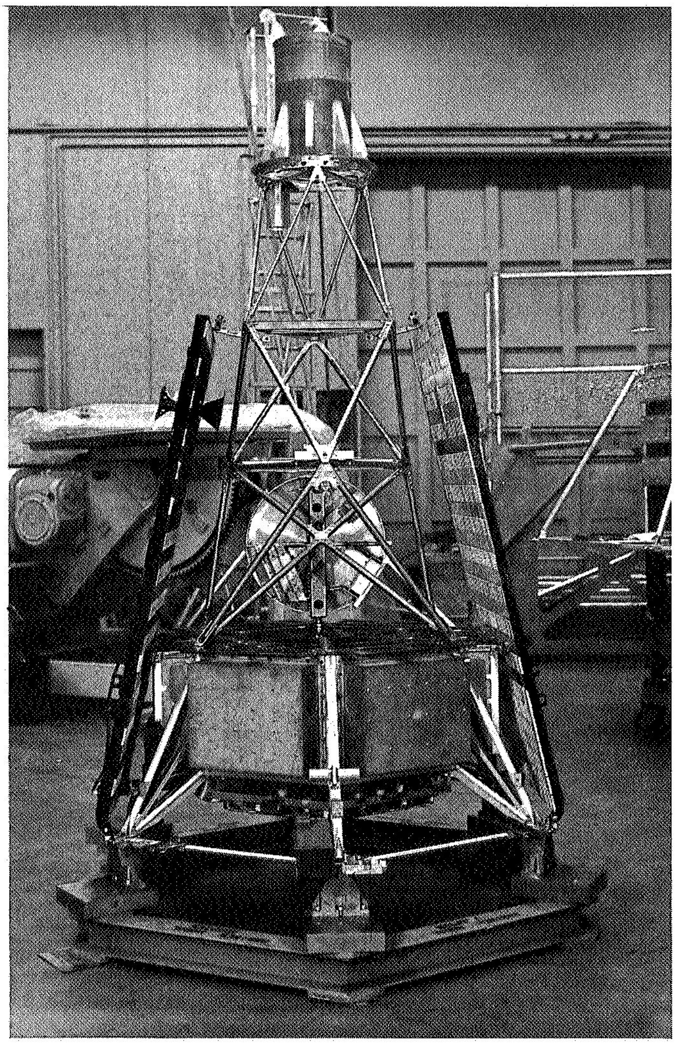


Fig. 9. ***Mariner 1*** **flight design structure**

The two most significant tests performed on the structural test model were the modal survey test and the structural qualification vibration test. The modal survey test setup is illustrated in Fig. 10. This test was accomplished in a manner similar to the solar panel modal survey test discussed earlier; that is, the various normal modes of the structure were excited by placing up to six small 25-lb shakers at points of significant motion in each of the modes. By this method, the first half-dozen frequencies and mode shapes were tabulated. After each mode shape was tabulated, the power was removed from all shakers simultaneously and an oscillograph record was made of the ensuing decay of several accelerometers scattered around the structure.

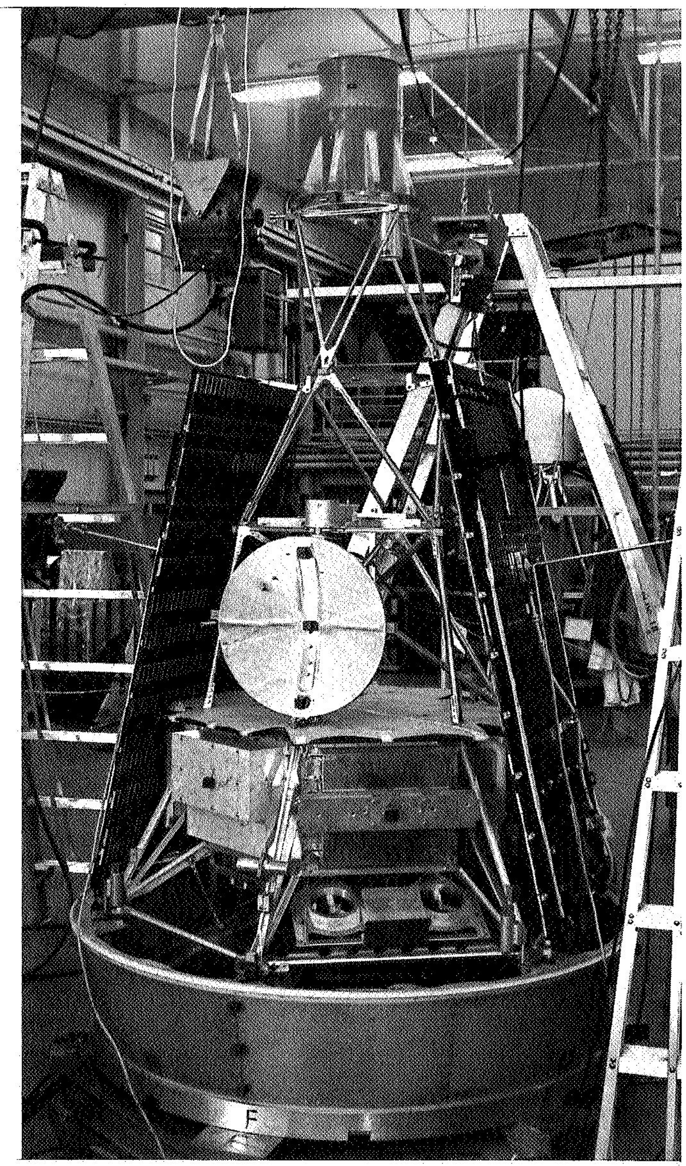


Fig. 10. ***Mariner 1*** **modal survey test setup**

Next, using a given spacecraft mass distribution and all the data accumulated from the modal survey, we could predict the response of the structure to a sinusoidal forcing function applied at the base of the spacecraft. This was accomplished with the aid of a computer program written for the JPL IBM 7090 computer. Thus we could estimate the behavior of the spacecraft when excited on the big shake table or, better yet, during flight.

Following the modal survey, the structural test model was submitted to the qualification vibration test. This test consisted of vibrating the spacecraft on a large 25,000-lb exciter and was intended to simulate the maximum expected flight vibration environment. During this test, the spacecraft was vibrated in each of three orthogonal directions. Figure 11 shows the spacecraft and exciter in the configuration for the shake in the plane of the solar panels. This particular mode of vibration was particularly interesting in that the top of the solar panel underwent a rather severe torsional motion at about 25 cps. This motion was started by the vertical motion of the lower solar-panel hinge points causing a rigid-body motion of the panels in their own plane, which then resulted in a rotation of the panels about their upper support point.

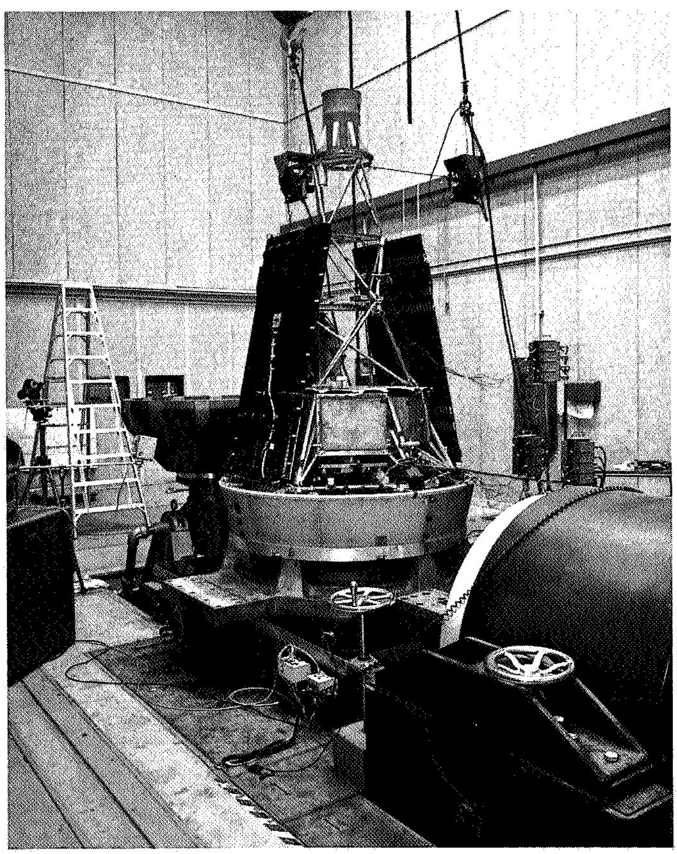


Fig. 11. Setup for qualification vibration test

After each part of the qualification vibration test, we attempted to excite the first few normal modes of the spacecraft with two of the modal survey shakers. Figure 11 shows these shakers as they were attached to the spacecraft after one part of the vibration test. Any failure in the spacecraft structure during the vibration test resulted in significant changes in the modal frequencies of the spacecraft. For example, a rivet failure on one of the solar-panel support links resulted in such a large shift of the first bending mode that we were unable to successfully isolate it.

To summarize, we might note that we have attacked the problem of spacecraft structural design, particularly as it applies to the spacecraft as a whole, in three different ways. First, we tried to make some sort of analysis. Second, we performed a modal survey test which resulted in the prediction of the response of the structure to a vibration load. Finally, we performed a vibration test with the spacecraft excited to an input vibration level simulating the flight environment. A good correlation between the results of these three approaches to structural design enhances the degree of confidence which we have attained in our structure.

OPEN DISCUSSION

Question: How closely could you predict some of these loads with your theoretical analyses?

Answer: In the case of the superstructure, which was a simple straightforward problem, we came out quite close. I think the first bending mode of the entire structure came out about 38 cps in real life, and we predicted something like 30 or 32 cps.

Question: How well could you predict damping?

Answer: Reasonably well, primarily because we had experience with most of the structures that we were using. In order to make our force-vibration analysis, we really didn't have to know the damping coefficients, as such. All we had to do was to assume a gain at some point in the structure, and we were somewhat familiar with how these structures behaved. It came out reasonably close.

Question: Did you say that in the analysis the solar panels were assumed uncoupled from the superstructure?

Answer: When we started we assumed they were uncoupled, and it turned out that when we finished, they were, indeed, almost uncoupled.

Question: What did you mean when you mentioned space frames in general?

Answer: A space frame is essentially a three-dimensional truss. We analyzed a space frame by assuming it to be a series of mass points connected by weightless members.

Question: Did you prefer to use welding or riveting?

Answer: The spacecraft superstructure and bus are all riveted. We had already learned from an earlier program that welded space-frame structures were not really safe. From an ease of fabrication, and so forth, welding is very nice; but every time we've gotten down to the wire and have had to make a decision of which way to go, it always has been determined that the riveted structure is much better. I think we made a good choice.

A Summary of the Temperature-Control Problems of Interplanetary Spacecraft

DONALD W. LEWIS

Supervisor, Temperature-Control Group
Spacecraft Development Section

In today's discussion, I am going to avoid most of the problems in temperature control, because there are just too many of them. Many of the thermal problems for an interplanetary spacecraft have been previously encountered in the development of Earth-satellite and lunar vehicles. The thermal environment during the prelaunch and boost phases of a trajectory and the effect of the Earth, the Sun, and the atmosphere on a low-orbiting vehicle is comparable, whether the vehicle is eventually boosted into an interplanetary trajectory or not. There are also thermal problems associated with the entry of a payload into a planetary atmosphere which are similar to those experienced on capsules re-entering the Earth's atmosphere. It is not intended to treat these problems in this discussion. They are only mentioned to indicate the scope of thermal design for a vehicle with a planetary mission.

In addition to these considerations, the cruise period of the trajectory presents some new problems which are peculiar to the interplanetary missions. The flight time to the nearest planets extends over periods of months. This allows adequate time for very small heat-flow rates to affect the equalization temperature. With long flight periods, particularly on trajectories toward the Sun, the properties of some materials change, due to prolonged exposure to direct solar radiation. However, the most pronounced effect on the design of an adequate temperature-control system is caused by a change of the intensity of the solar radiation.

The intensity of the solar radiation at Venus is approximately twice that of its intensity near Earth. To illustrate how this affects the temperature control of a Venus probe, consider the temperature limits of a typical component as illustrated in Fig. 1. This represents a solar-dependent assembly. A temperature balance between the energy that is absorbed by the sunlight and that emitted by this assembly must be met. It has a survival temperature of −40 to +160°F. The survival temperature limits are set on the basis of permanent damage to the component, and the operating limits define an acceptable range of temperatures during operation of the equipment. Assume that the equipment is required to operate during the entire transit from Earth to Venus, and that its temperature limits during this period are 0 to 120°F. While going to Venus the solar intensity just about doubles. In the case where the temperature of the assembly is solar-dependent, the temperature rise due to the change in solar intensity would be only approximately 90°F. This requires that the temperature of the assembly when leaving the Earth's vicinity must be predicted to within ±15°F.

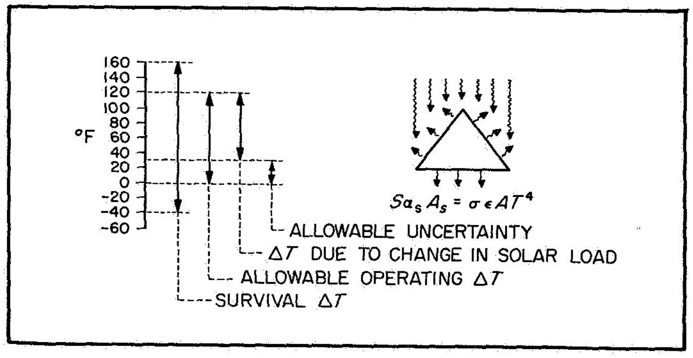


Fig. 1. Solar-dependent item

It may be possible to make the temperature of the assembly less solar-dependent if this small tolerance is recognized as a problem early enough in the design. Solar independence is difficult to achieve completely. Unfortunately, nonthermal constraints such as power limitations, configuration constraints, and in some cases, the viewing requirements of the assembly itself prevent this approach.

Often it is desirable to group a number of assemblies into a solar-independent group to take advantage of the

resulting higher packaging efficiency. By grouping assemblies, it is possible to take advantage of the heat dissipated by an active assembly to keep an inactive assembly above its lower survival limit. Precautions must be taken to package the high-heat-generating assemblies in such a way that an adequate radiating area exists and that a minimum number of mechanical joints exist between the heat source and its radiator. Conversely, precautions must be taken to minimize the heat losses from assemblies which generate very little heat. Whether packaged in groups or singly, the thermal losses must be maintained by the heat dissipated with the assembly.

The operating temperature limits of a group of assemblies is not usually as wide as the limits on each assembly. In Fig. 2, I have sketched an insulating blanket which shields this particular part from the Sun. If the part is made completely independent of the Sun, it will eventually get too cold during the three-month period that it takes to get to Venus; therefore, it is necessary to rely on the internal power that might be dissipated to keep it from getting too cold. Assemblies "A" and "B," both thermally connected, are illustrated here. Assume that assembly "A" has an operating range of 40 to 160°F, and assembly "B" has a range of 0 to 120°F. While both of these assemblies have an operating range of 120°F separately, an acceptable range of only 80°F exists when they are thermally grouped together, or a tolerance of ±40°F.

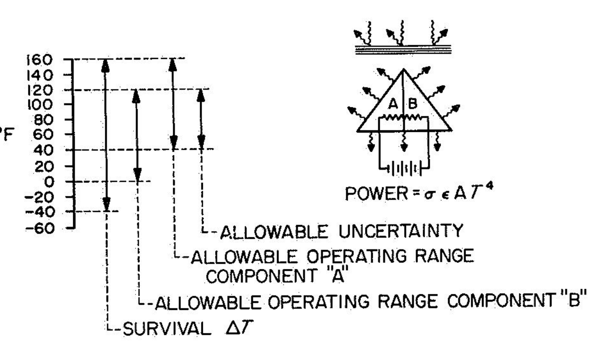
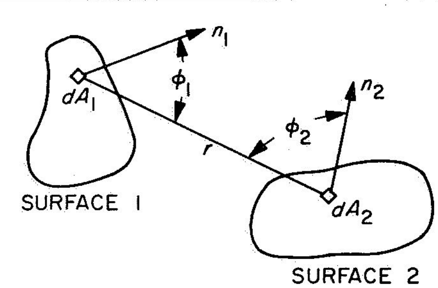


Fig. 2. Solar-independent item

Whether a spacecraft is Sun-dependent or independent, or composed of items in each category, the temperature must be predicted within allowable tolerances. The engineer has at his command a variety of analytical and experimental tools which are available to make these predictions and on which to base the design of a temperature-control system.

The accuracy of the analytical solution to the heat-transfer problems encountered is not as limited by an understanding of the physics involved as by analytical techniques available. Some of the limitations come from uncertainties in empirical factors such as total absorptance and emittance, but the most serious limitation lies in the definition of the boundary conditions of a problem. The simplifying assumptions that are required to mathematically state the problem often introduce more error than can be allowed in their solutions. Figures 3 and 4 have been included to demonstrate the difficulty of analytically stating the problems encountered.[1]

Figure 3 defines a term called a form factor, which is a measure of the percent of radiant energy exchanged between two radiators. It is typical of the type of problem usually found in the first chapter of any good book on radiation heat transfer, from which a student is usually assigned homework to determine the form factor between two finite planes.

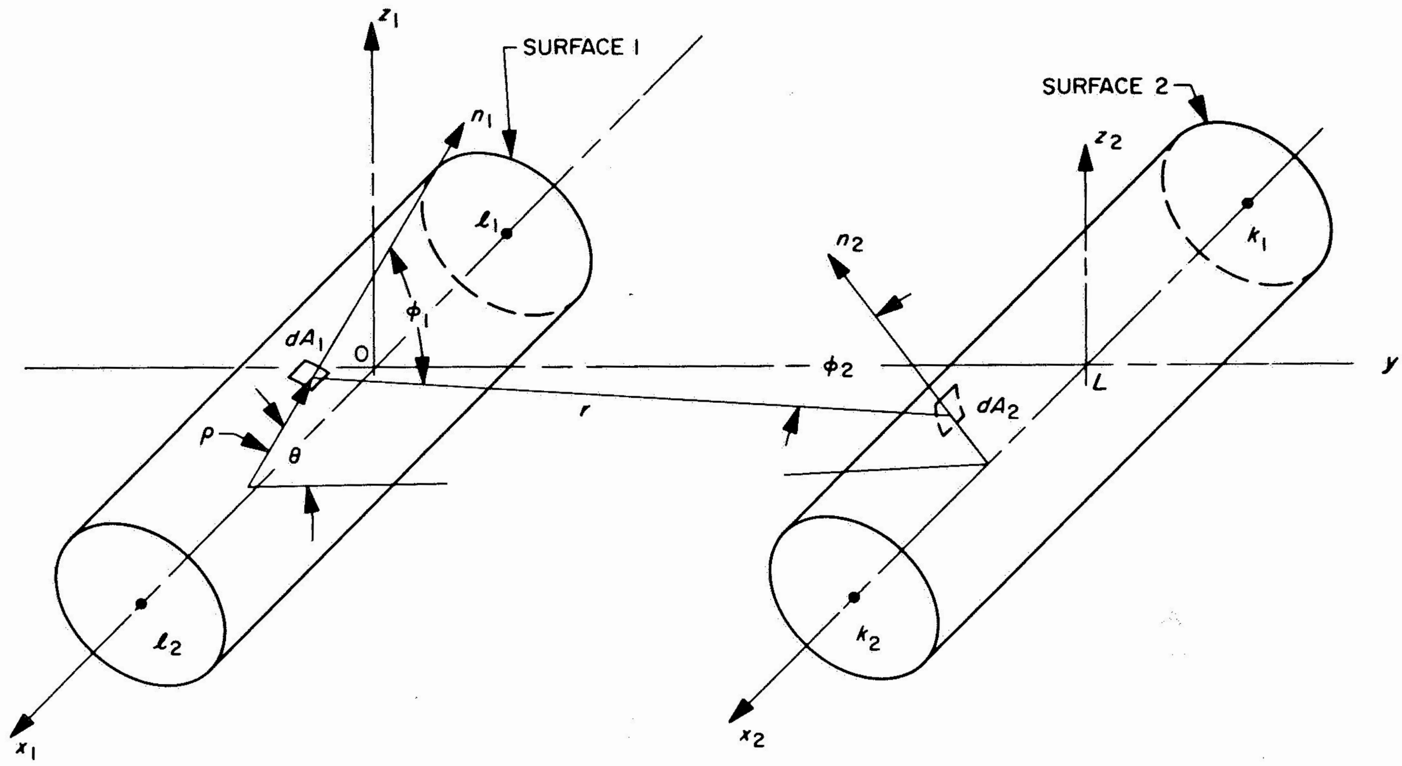


$$F_{1\text{-}2} = \frac{1}{\pi A_1} \int_{A_1} \int_{A_2} \frac{\cos\phi_1 \cos\phi_2}{r^2} \, dA_1 \, dA_2$$

Fig. 3. Radiation form factor—general expression

A more difficult problem, illustrated in Fig. 4a, is the determination of the form factor between two parallel cylinders. All of the various terms, angles, and dimensions must be defined. In spite of the rather simple geometry involved, the analytic expression (Fig. 4b and 4c) is rather complex. This type of problem is well adapted to solution on digital computers; however, the inherent weakness of the analytical approach can be easily recognized when typical views of a spacecraft (Fig. 5 and 6) are compared to the assumed idealized surface. Fortunately, in the case of the form factor, an optical tech-

[1]The material shown in Fig. 4 was taken from *Numerical Determination of Radiation Configuration Factors for Some Common Geometrical Situations,* Plamondon, J. A., Technical Report No. 32-127, Jet Propulsion Laboratory, Pasadena, July 1961.



a. For parallel cylinders

$$A_1F_{12} = \int_{l_1}^{l_2}\int_{k_1}^{k_2}\int_{\cos^{-1}(\rho+\zeta)/L}^{\cos^{-1}(\zeta-\rho)/L}\int_{\theta_1}^{\theta_2} F(x_1, x_2, \psi, \theta)\, d\theta\, d\psi\, dx_2\, dx_1$$
$$+ \int_{l_1}^{l_2}\int_{k_1}^{k_2}\int_{-\cos^{-1}(\zeta-\rho)/L}^{-\cos^{-1}(\rho+\zeta)/L}\int_{\theta_3}^{\theta_4} F(x_1, x_2, \psi, \theta)\, d\theta\, d\psi\, dx_2\, dx_1$$
$$+ \int_{l_1}^{l_2}\int_{k_1}^{k_2}\int_{-\cos^{-1}(\rho+\zeta)/L}^{\cos^{-1}(\rho+\zeta)/L}\int_{\theta_2}^{\theta_3} F(x_1, x_2, \psi, \theta)\, d\theta\, d\psi\, dx_2\, dx_1$$

b. Analytic expression

$$(x_1, x_2, \psi\ \theta) = \frac{\zeta\rho}{\pi}\ \frac{[L\cos\theta - \rho - \zeta\cos(\theta+\psi)]\,[L\cos\psi - \zeta - \rho\cos(\theta+\psi)]}{[(x_1 - x_2)^2 + (L - \rho\cos\theta - \zeta\cos\psi)^2 + (\rho\sin\theta - \zeta\sin\psi)^2]^2}$$

$$\theta_1 = -\cos^{-1}\frac{\rho - L\cos\psi}{\zeta} + \psi$$

$$\theta_2 = -\cos^{-1}\frac{\zeta}{(\rho^2 + L^2 - 2L\rho\cos\psi)^{1/2}} + \tan^{-1}\frac{\rho\sin\psi}{L - \rho\cos\psi}$$

$$\theta_3 = \cos^{-1}\frac{\zeta}{(\rho^2 + L^2 - 2L\rho\cos\psi)^{1/2}} + \tan^{-1}\frac{\rho\sin\psi}{L - \rho\cos\psi}$$

$$\theta_4 = \cos^{-1}\frac{\rho - L\cos\psi}{\zeta} - \psi$$

c. Analytic expression

Fig. 4. Radiation form factor

nique[2] using an instrument called a form factometer (Fig. 7) has been recently developed, which greatly simplifies its evaluation. In like manner, thermal analysis is benefiting from research in the surface properties of materials and in studies of conduction across structural joints under vacuum conditions. If the roles of analysis with its limitations are recognized during the preliminary design of a spacecraft, it may be possible to shape the configuration in such a way that it can be described analytically.

[2] *Determination of Radiation Configuration Factors,* Hickman, R. S., Technical Report No. 32-154, Jet Propulsion Laboratory, Pasadena, December 1961.

Complexity is not a new problem in engineering. A long-accepted approach that can yield useful hardware without a rigorous analysis is the cut-and-try technique. The degree to which the experimental technique is valid is closely linked to the fidelity of the test environment. While the exact simulation of the space environment is not possible, it has been demonstrated that for heat-

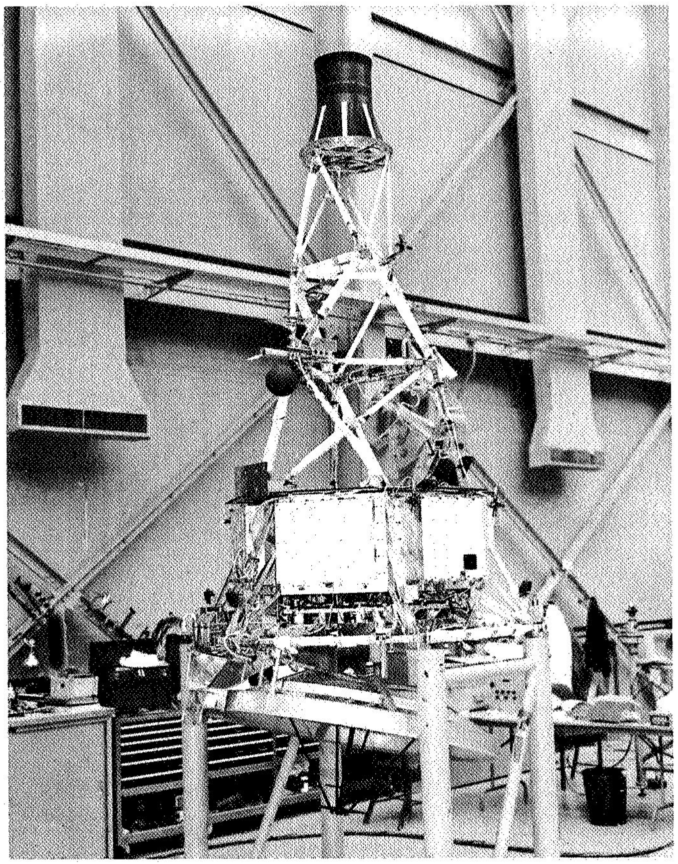


Fig. 5. Typical interplanetary spacecraft

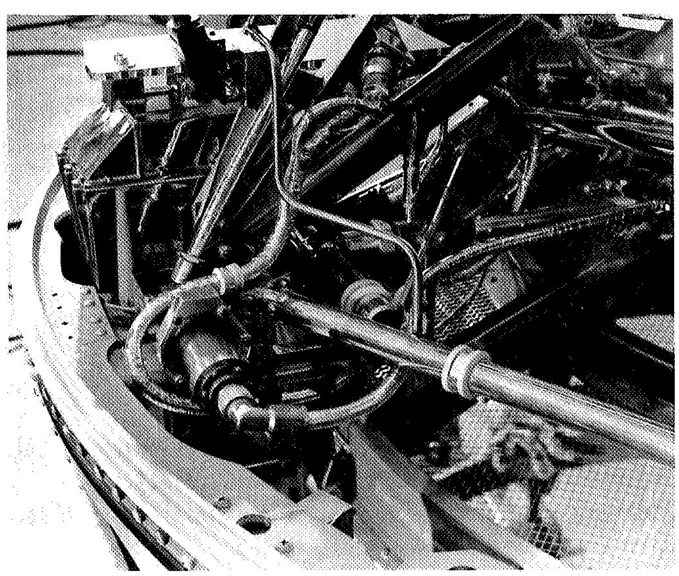


Fig. 6. Detail of a portion of a spacecraft

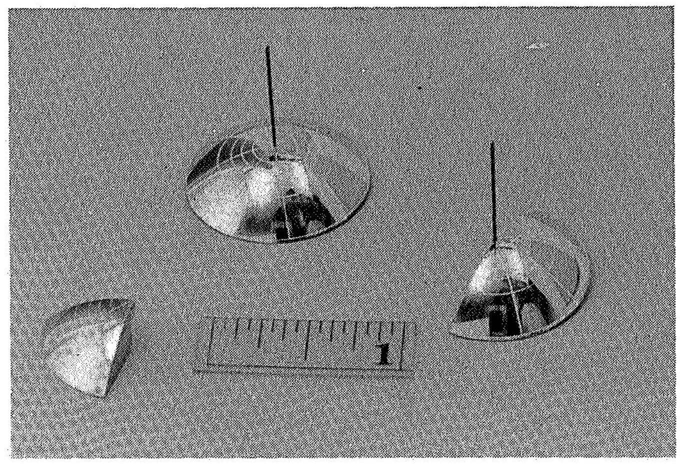


Fig. 7. Radiation form factometer

transfer studies, the vacuum and heat-sink properties can be adequately approximated. For a design that does not depend on the solar radiation for its temperature control, the cold-wall vacuum chamber is a very powerful experimental tool. Even for a design that is solar-dependent, techniques using radiant or resistance heaters give results comparable to the accuracies of analysis.

Figure 8 is a photograph of such a test setup using a cold-wall vacuum chamber with a full-size thermal model of a spacecraft. In this case, the energy that would be absorbed from the sunlight was supplied by resistance heaters. The thermal model is structurally just like the spacecraft (Fig. 5) with all the conduction paths the same. Resistors are used instead of electronic components. This vacuum chamber was provided with a nitrogen-cooled shroud or heat sink. As far as the spacecraft is concerned, it does not know whether it is dumping heat to the infinite heat sink of space or to the very cold black walls of the vacuum chamber.

An important requirement for a realistic evaluation of a spacecraft in a vacuum chamber is an adequate simulation of sunlight. There are problems in trying to extrapolate from a cold-wall test to the solar environment. The spectrum of sunlight has been checked a number of times, on tops of high mountains, and in high rocket flight, but the simulation of this lighting is most difficult. For in-

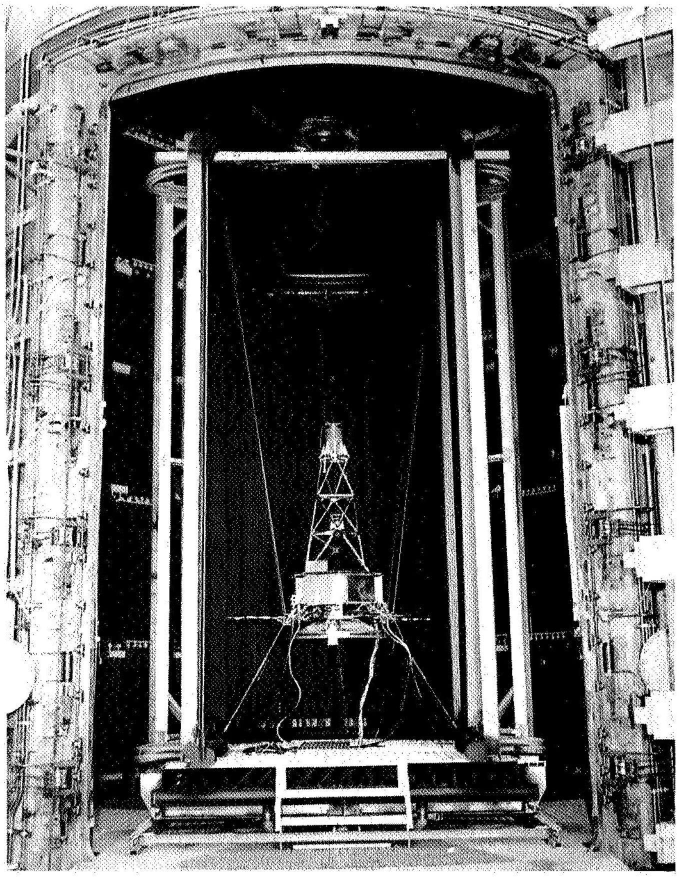


Fig. 8. Large cold-wall vacuum chamber

stance, one arc lamp may be used because its spectrum is not too far off, but it is hard to work with; or another arc lamp may be used because it is easier to work with, but then its spectrum may be more in error. Only when more accurate simulation of the solar radiation becomes available will more valid preflight testing be accomplished. Unfortunately, this cannot be assured at this time, for in the fabrication of such a simulator many compromises must be made in the intensity, the collimation, and the spectrum of the illumination. Care must be taken that the correction factors for these deviations of simulators from the actual solar radiation do not introduce more uncertainty than present techniques.

Thermal control is a very challenging subject. It is one in which engineers can make the greatest contributions. Only ten years ago, radiation heat-transfer people were worrying about the amount of energy transferred from a fire box to boiler tubes. If we knew the amount of heat transferred within 30%, we would consider that to be a good calculation. Now we are called upon to hold temperatures within about 3% variation with the same analytical and experimental tools. Granted that in spacecraft there are internal conduction problems, it is ultimately a radiation problem that defies solution. How do we get from a 30% type of theory or technique to 3% and do it before Venus gets here? You who are in school will not be working on this Venus opposition, but maybe some of you will be working on thermal problems for the next Venus opposition in which you will be backed up against the wall and be expected to produce a certain degree of temperature control with tools that are too crude. There is so much to be learned and to be developed.

Neither the analytical nor the experimental approach to the thermal problem is adequate in itself. Furthermore, the combination of the best analytic and experimental methods available today results in only marginal assurance that the temperature of an assembly can be controlled. It is important that these limitations be recognized so that the spacecraft is designed with realistic and consistent reliability. It is not reasonable to demand that an electronic circuit have a 3-σ probability of successful operation while at the same time its temperature cannot be assured to be within its known operating limits.

I have talked a lot less about temperature control today than the subject would warrant. I have talked about only one phase of it. Now, some of you may have questions.

OPEN DISCUSSION

Question: Why do you use gold plating on a spacecraft, Sir?

Answer: Let us call the surface that is in the sunlight A_s (A sub Sun). It will receive solar energy, and the energy that hits it is either reflected or absorbed. The percent that is absorbed is called the absorptance. That which is absorbed raises the temperature until such time as energy emitted is equal to the amount that has been absorbed. But, what is that temperature? This depends on the surface properties of the material.

I have two plates here. One is gold plated and the other is polished aluminum. They have a very high reflectance and are such good mirrors that you could shave by them if you wanted to. Having a very high reflectance, they would absorb very little. Actually, they absorb only about 25% of the energy intercepted by them. At the same time, their emissivity is very low—only 5%. So if this gold plate were in the sunlight, it would get very hot. In the vicinity of

Venus, it would get to be about 800°F. A little black spot of paint on the surface would change the whole picture.

There are two other surfaces that I would like to show you. These two (a gold plate and a white plate) are just alike so far as their solar absorptance is concerned; both of them absorb about 25% of the incident energy. But this one, the white plate, has a very high emissivity, so if we have a requirement for dumping a lot of heat and absorbing very little, we paint it white. Most of our sunlit surfaces are white, with the exception of those that are polished. Both of them are to absorb a minimum of energy. This one dumps it (the white one), and this one retains it (the gold one). This plate (a black one) is at the other end of the scale, so far as solar absorptance is concerned. It absorbs a great percentage of the solar energy that hits it—about 85 to 90%. Its emissivity is comparable to the white, at room temperature. These are the tools that we work with.

Question: Do you ever use a venetian-blind-type of device?

Answer: Yes, the spacecraft that we showed you has this type of device, and its purpose is to change the emissivity from something small to something large. In a box on which it is placed, there may be a varying number of watts dissipated as electrical energy, which must be conducted to some radiating surface and radiated out into space.

When the louver is open it can radiate more energy because a higher emissivity surface is exposed. When it is closed the louvers act as a radiation shield, blocking the energy radiated to space.

Some Materials Problems in Spacecraft

RALPH A. HAPPE

Supervisor, Materials and Methods Group
Applied Mechanics Section

It is convenient to divide materials problems associated with the design of spacecraft into two general categories: (1) those problems which are *not* unique because of the space environment; (2) those problems which *are* unique because of the space environment.

Because of limited time, I will devote most of this lecture to a discussion of the second category. Before proceeding, however, I'd like to touch briefly upon some of the problems in the first category.

The problems associated with the design and fabrication of minimum-weight structures have been faced with varying success by the aircraft industry for years. In order to keep weight to a minimum it is necessary to design with very low margins of safety. Superimposed on the problem of working with low margins of safety is the problem of dealing with difficult-to-fabricate materials such as magnesium, aluminum, titanium, and stainless steel. The fact that low margins are used dictates that fabrication procedures be much more exacting than those for Earth-bound structures.

Pressure vessels are examples of such practices. Normal practice for non-flight pressure vessels is to design to a burst-to-operating pressure ratio of about 4. Flight vessels, on the other hand, are designed with burst-to-operating ratios of 1.2–2.2. The higher ratio is used when the vessel must be pressurized fully in the vicinity of personnel. The lower end of the range is used when pressurization can always be done away from personnel.

The balance of this lecture will be concerned with those problems that are unique to space environmental aspects of materials usage. I will touch on the subject of metals and organic polymers and then go into more detail on white paints.

Figure 1 presents a compilation of our present knowledge of the space environment. From an engineering point of view we are learning today to deal intelligently only with the first three items: pressure, temperature, and electromagnetic radiation. We need much more knowledge of solid particles, neutral gas, ions, and electrons before we will be in a position to give full recognition to these factors in the design of our spacecraft.

Let us return to the first three items. Present technology is on the threshold of learning how to achieve pressures in vacuum chambers approaching those believed to exist in outer space. Fortunately, from a materials selection point of view, much useful data can be obtained at more reasonable pressure, say 10^{-6} to 10^{-8} mm of mercury.

Spacecraft temperatures are held in a fairly reasonable range for most common engineering materials. This is dictated by the requirements of electronics gear carried aboard. Since the spacecraft is protected from the atmosphere on launch by a shroud that is later kicked off, we do not have a general problem with very high temperatures.

In the future we will undoubtedly be dealing with much lower temperatures, especially for propellant tanks for high-efficiency propellants such as liquid oxygen or liquid hydrogen. At present, and for the next few years, we do not have to be particularly concerned with this aspect in the design of our spacecraft.

I will forego further comment on electromagnetic radiation since it will be covered in more detail as we talk about temperature-control surfaces, where it is an extremely important factor.

Except in the portion of this talk on temperature-control surfaces or white paints, we will be considering only the environmental effects of temperature and pressure.

Pressure	10^{-11} to 10^{-14} mm of Hg
Temperature	Dependent on vehicle requirements (for Ranger – T = 35°C ± 15°C) (maximum excursion = ± 100°C)
Electromagnetic Radiation	Solar constant = 2.0 cal/cm^2–min 93% between 0.3 μ and 2 μ Black-body temperature, approximately 6000°K
"Solid" Particles	
Meteoroids (> 1mm radius)	v = 15 mi/sec d = 0.1 gm/cm^3 Flux = 2.5×10^{-12} particles/cm^2–sec
MicroMeteorites (< 1mm radius)	v = 6 mi/sec d = 0.1 gm/cm^3 Concentration = 10^{-14} to 10^{-15} particles/cm^3
Neutral Gas (hydrogen)	Concentration < 1 atom/cm^3
Ions	
Mev Range	Flux = 4 particles/cm^2–sec (bursts to 120 particles/cm^2–sec)
1 Kev to 1 Mev	Flux = 10^9 – 10^{12} particles/cm^2–sec
Thermal Ions (below 1 Mev)	(none above 60,000 mi)
Electrons	Concentration = 100 particles/cm^3 (as high as 10^6 particles/cm^3 in Val Allen Belts, 600 – 60,000 mi)

Source: Materials Problems Associated with the Thermal Control of Space Vehicles – MAB-155-M

Fig. 1. The space environment

METALS

A first approximation of the behavior of metals in a temperature-vacuum environment can be obtained through the use of the well-known Langmuir equation, which is shown in Fig. 2. This equation is quite useful for predicting the sublimation rate of elemental metals. It is generally applicable to pressures of about 10^{-5} mm of mercury or lower, where the mean free path is sufficiently long that atom-to-atom collisions can be ignored. For this reason, of course, there is no expression for ambient pressure in the Langmuir equation. The Langmuir equation was originally developed in connection with evaporation of filament materials for light bulbs at very high temperatures.

There is some risk, of course, in using the Langmuir equation at lower temperatures, because vapor pressures for many elements must be extrapolated. This seems to work pretty well for most elements. There is considerable question at present, however, as to how accurate the calculation is for magnesium, a metal we are very much concerned with. Direct experimental data on evaporation of magnesium in laboratory chambers indicates that the sublimation rate of magnesium is considerably less than one would predict by using the Langmuir equation and extrapolated vapor pressures. We know of no case where the actual rate of evaporation is higher than that predicted by the Langmuir equation. Therefore, we can consider it a useful device for obtaining the upper limit of evaporation.

Figure 3 shows the elements arranged in order of decreasing vapor pressure. Magnesium, for most practical purposes, can be considered to be a borderline material for space applications. We attempt to keep to a minimum the amount of elements such as zinc and cadmium. It is of some passing interest to note that the elements lead and tin, which are constituents of common soft solder, are really quite stable, contrary to popular conception. Most common structural materials such as aluminum, stainless steel, and titanium have very low vapor pressures. One can show with the Langmuir equation that evaporation

$$p = K_1 w \sqrt{T/M} \qquad \text{or} \qquad R = \frac{K_2 p}{d \sqrt{T/M}}$$

p = Vapor pressure, mm of Hg

K_1 = 17.14

w = Rate of evaporation, gm/cm^2-sec

T = Temperature, °/K

M = Molecular weight

R = Rate of evaporation, cm/yr

$K_2 = 1.8 \times 10^6$

d = Density, gm/cm^3

assume gaseous molecules monatomic

except

Se, Te, Sb, Bi, C, which are assumed diatomic

Fig. 2. The Langmuir equation

Highest	Xe	Zn	Ag	Ti	
	Br	Te	Sn	U	
	I	Mg	Al	V	
	Hg	Sr	Be	Rh	
	S	Li	Cu	Pt	
	Cs	Sb	Au	B	
	Rb	Ca	Ge	Ir	
	K	Ba	Cr	Zr	
	P	Tl	Fe	Mo	
	Cd	Bi	Pd	C	
	Na	Pb	Co	Re	
	As	In	Ni	Ta	
	Po	Ga	La	W	Lowest

Fig. 3. Elements arranged in order of decreasing vapor pressure

rates of these metals will be negligible even for extremely extended time periods at the moderate temperatures with which we are presently concerned.

So far, we have been talking primarily about the effect of evaporation on the metal itself. This problem is not very difficult to treat analytically. There is another problem which must be considered—evaporation of small quantities of the metal and the subsequent deposition in unwanted areas such as across open switches, or on insulating boards, with possible resultant short circuits. The practical approach to this problem is to use, where possible, metals with low vapor pressures. Critical geometries should be carefuly tested in vacuum to determine whether this problem will manifest itself.

POLYMERS

I am sure all of you have, at one time or another, looked into your television sets and noticed the large amounts of plastics (or polymers) that are used in electronic hardware. In spacecraft design we have, basically, a mass of electronic gear held together with structure. Polymers are used as insulation for wires, as potting compounds, as circuit boards, and in a host of other applications where their low thermal and electrical conductivity, and relative ease of forming indicate their usefulness. Polymers are also used to a lesser extent in structural or semistructural applications where one portion of the spacecraft must be thermally insulated from another. There are other important applications of polymers where their radio-frequency transparency is important. An example of this is the rods that support the ground plane of the high-gain antenna. These rods are made of fiberglass-reinforced plastic.

So much for the engineering applications of polymers. We will now briefly consider the changes that can occur to such materials during service in a space vehicle. Because of their structural complexity, polymers do not lend themselves to a simple analytical treatment, as was the case for metals. Deterioration of polymers is a field of study in itself. At the risk of over-simplifying, we can broadly break down the mechanisms of concern into two categories: (1) vacuum-temperature effects, and (2) temperature effects. An example of the first is the detachment of low-molecular-weight fractions from the basic polymer chains. Such structural changes can alter properties with or without loss of the fragments that are detached. In addition, low-molecular-weight additives, such as plasticizers, can evaporate with or without significant changes in properties. An example of the second category is modification of the basic chain structure by scission or by cross-linking. Such changes need not be accompanied by weight changes to be important.

The mechanisms suggested in the preceding paragraph can manifest themselves in a number of ways with regard to engineering properties. Some of the changes that can occur are: (1) embrittlement and/or stiffening, (2) shrinking, (3) lowering of strength, and (4) alteration of electrical and physical properties.

In addition to the effects on the polymer itself, we must also consider the effect of deposition of material that may be given off. This is a situation that is analagous to the problem I mentioned in connection with metals. An example of this latter phenomenon is given in Fig. 4, which shows a gamma-ray spectrometer that had been subjected to a solar-simulation test under a vacuum of about 10^{-5} mm. Note the dark spot in the center of the white strip on the left side of the sphere. This spot was of some concern because in the area of the dark (actually brown) marking the optical properties of the white strip, placed there for temperature-control reasons, are markedly changed. At first it was suspected that the white paint had yellowed during the test. A closer examination proved that the brown spot resulted from exchange of material from the insulation in the cable leading into the spectrometer to the white surface.

Figure 5 shows a different view. At about the center of the picture you can see the discoloration of the cable itself. A special test was devised to determine what part

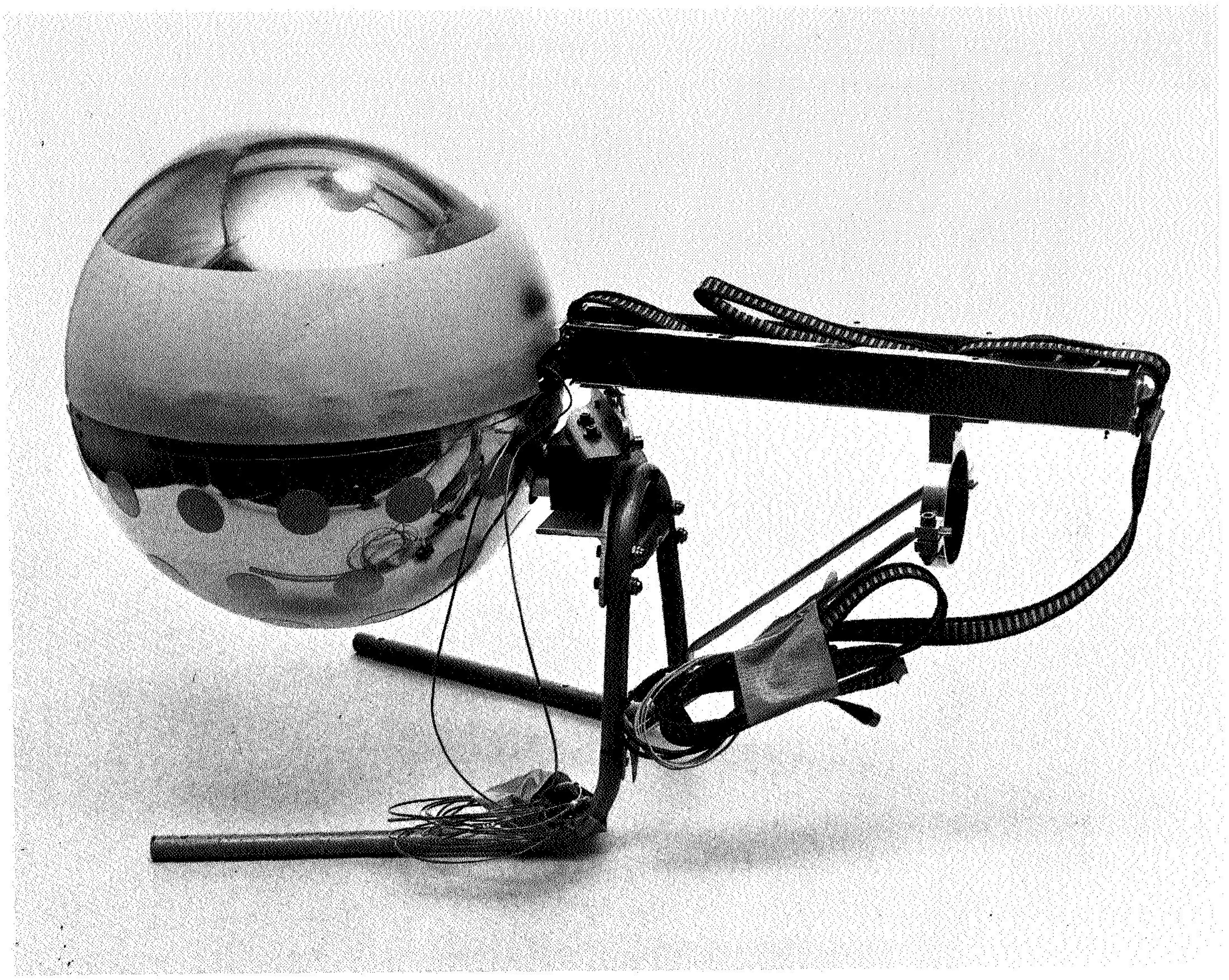


Fig. 4. Gamma-ray spectrometer after solar-simulation test

of the cable was giving off material. A small aluminum box was constructed which contained pertinent parts of the cable. At one end of the box was a slit. Placed opposite the slit was a polished aluminum plate, half of which was painted white. All this was placed in the vacuum chamber and the box was heated by infrared radiation.

Under the test conditions, the co-ax cables caused staining of the plate, as did the outer jacket of the co-ax cable itself. The inner insulation proved to be relatively innocuous. The balance of the components of the cable did not cause any difficulty under the simulated test. Therefore, it was concluded that the culprit was the outer jacket (irradiated polyolefin) of the co-ax cable.

Of equal importance to the technical side of polymers and their behavior in space environment is what one might term the commercial side of the question. The polymer industry has extensively adopted the use of trade names to cover their products. Classifications are relatively unclear, and it is next to impossible to get reliable information as to the exact composition of many commercial products. In addition, relatively small quantities are required in spacecraft construction. Therefore, we do not have any particular wedge to obtain the information that is so vital to our applications. It is also very difficult to control the usage of the polymers because of the tremendous diversity of devices used and the numerous places that they are made. For example, every assembly technician seems to have his favorite potting

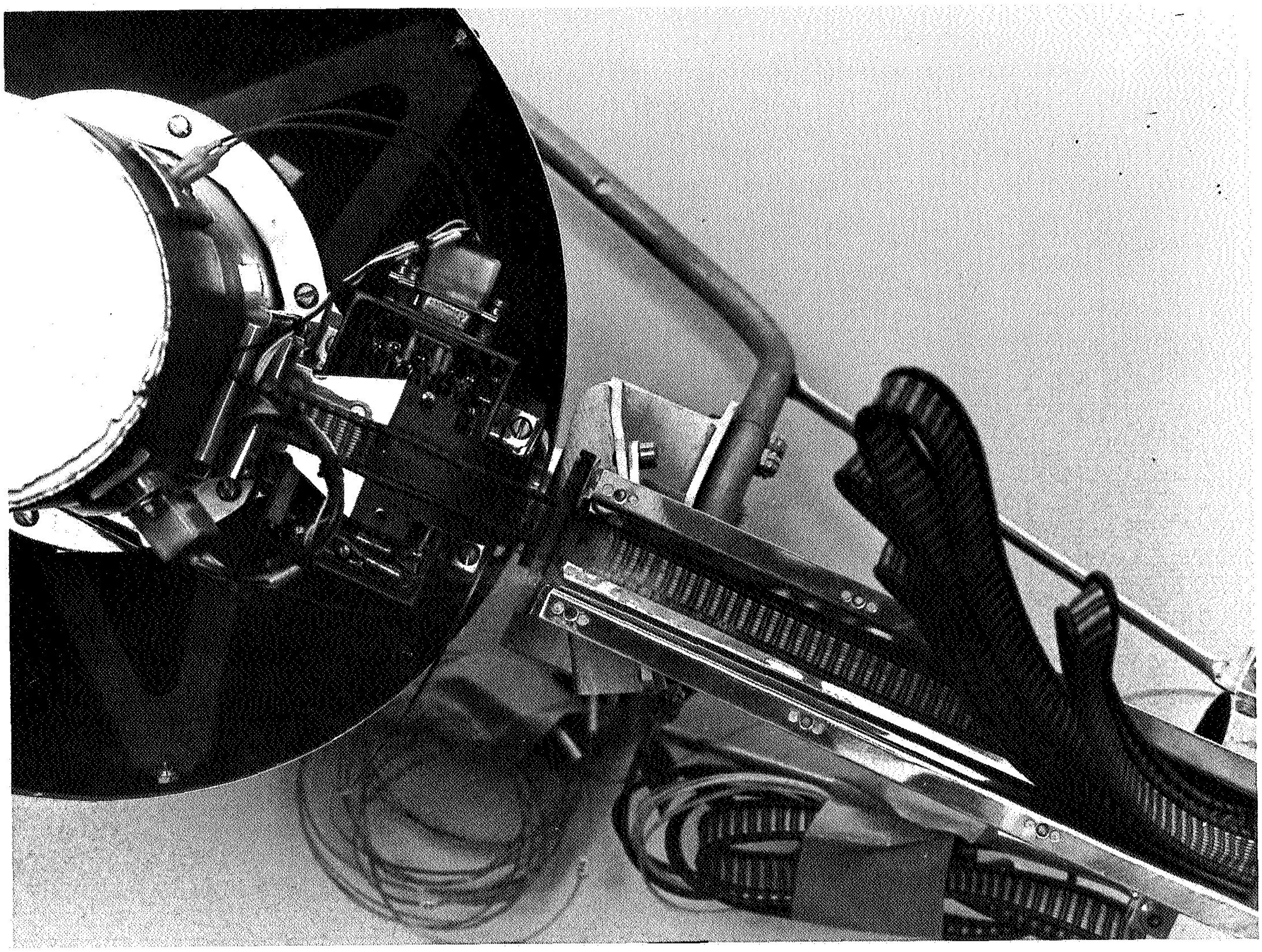


Fig. 5. Another view of the gamma-ray spectrometer

compound, which he prefers for one reason or another. The electronics industry has grown so rapidly that there is very little standardization.

WHITE PAINTS

In the previous lecture you were given some insight into the role of spacecraft surfaces in the over-all task of controlling temperature of the electronics. A number of surfaces are used to provide the necessary optical properties. Depending on the particular temperature-control requirement of the given area, the following classes of surfaces are used: (1) polished metals or polished electroplates, (2) aluminum paints, (3) black paints, and (4) white paints. All of these surfaces have characteristic ranges of optical properties expressed as solar absorptance and infrared emittance.

The balance of this talk will be concerned with the particular problems in connection with white paints. Such surfaces are usually exposed continually to the Sun during service. They have a relatively low solar absorptance, along with a high infrared emittance. Therefore, they tend to run cool while in the Sun. Such surfaces are used on the *Mariner 1* in the heat shield and on numerous portions of the spacecraft that are exposed to the Sun, either when it is Sun-oriented or during the tumbling phase.

Unfortunately, most surfaces that fall into the "white" category are very susceptible to darkening or yellowing as a result of ultraviolet radiation in vacuum. The degree of yellowing, of course, is a function of time. This problem can become quite severe on a Venus or Mars mission, where transit times range from 3 to 6 months. There are

no commercial paints that we know of that are capable of withstanding such exposure without severe darkening. Of course, such darkening would result in an increase in the solar absorptance and a drastic disruption of the thermal balance.

Before proceeding further, it is convenient to consider the nature of the electromagnetic radiation emanating from the Sun. Figure 6 shows the spectral distribution of solar radiation, both at the Earth's surface and above the Earth's atmosphere. This Figure illustrates an important feature of the extraterrestrial solar distribution. Note that the intensity of ulraviolet is much higher outside the Earth's atmosphere. This is because the Earth's atmosphere is an efficient filter for ultraviolet radiation. It turns out that the ultraviolet radiation in the region of from 0.2 to 0.4 μ is extremely damaging to white surfaces. Therefore, we have a problem that is quite different from what we are used to in ordinary Earth applications. In addition, the problem is aggravated by the lack of atmosphere, since oxygen frequently plays a role in reversing the process by the familiar bleaching reaction. We recognized early that much of the data available on ultraviolet degradation of white paints was virtually worthless for our applications, since most of the data have been obtained in air.

Figure 7 gives some recent data obtained from rocket soundings on the solar distribution in the shorter wavelength portion of the spectrum. Many scientists are concerned about the effects of the helium and hydrogen radiations. Preliminary tests to date by the National Research Corporation indicate that the effects of the very short wavelengths are not so severe as the effects of the 0.2 to 0.4 μ wavelengths. Therefore, a material that is suitable for use in a 0.2 to 0.4 μ region will probably also be satisfactory so far as the shorter wavelengths are concerned. In our work with white paints we intend to check the effects of shorter wavelengths on paints that appear promising for the 0.2 to 0.4 μ region.

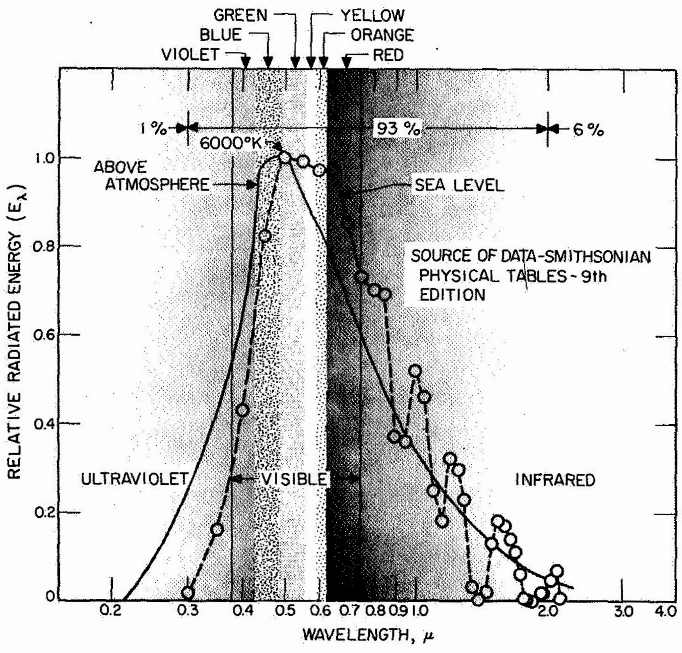


Fig. 6. Spectral distribution of solar radiation

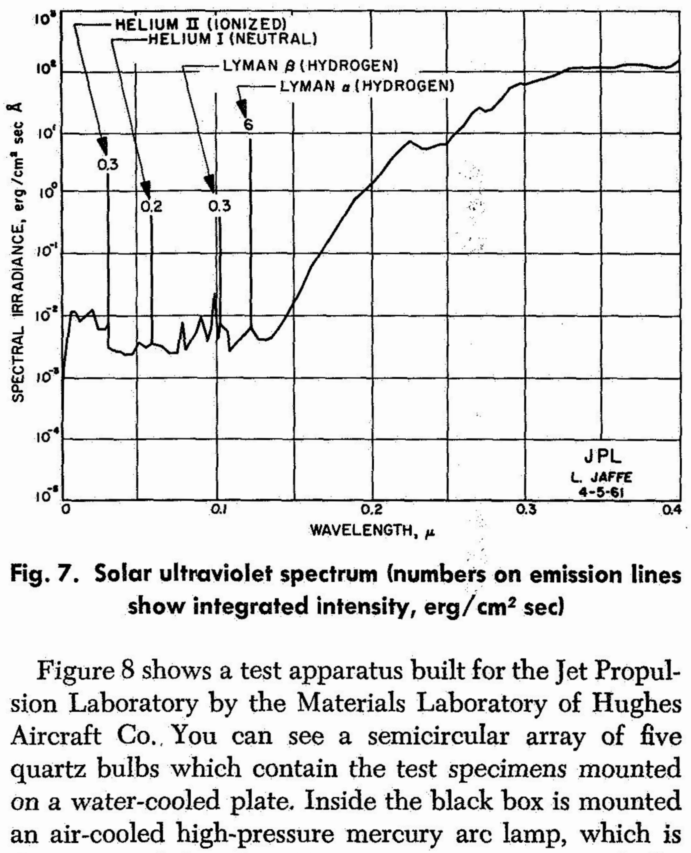


Fig. 7. Solar ultraviolet spectrum (numbers on emission lines show integrated intensity, erg/cm² sec)

Figure 8 shows a test apparatus built for the Jet Propulsion Laboratory by the Materials Laboratory of Hughes Aircraft Co. You can see a semicircular array of five quartz bulbs which contain the test specimens mounted on a water-cooled plate. Inside the black box is mounted an air-cooled high-pressure mercury arc lamp, which is the source of the ultraviolet. During exposure the bulbs are lowered into the box so that the paint sees the ultraviolet bulb. Test bulbs are, of course, evacuated during the run.

Figure 9 shows an apparatus for a similar purpose at Armour Research Foundation. Armour is doing work under a JPL subcontract to develop suitable white paints for long-time exposure to ultraviolet and vacuum for future spacecraft. The Armour version of the test apparatus uses a single chamber with a very high pumping rate. Three of the high-pressure mercury arc lamps are contained in the lid of the chamber (Fig. 10). The

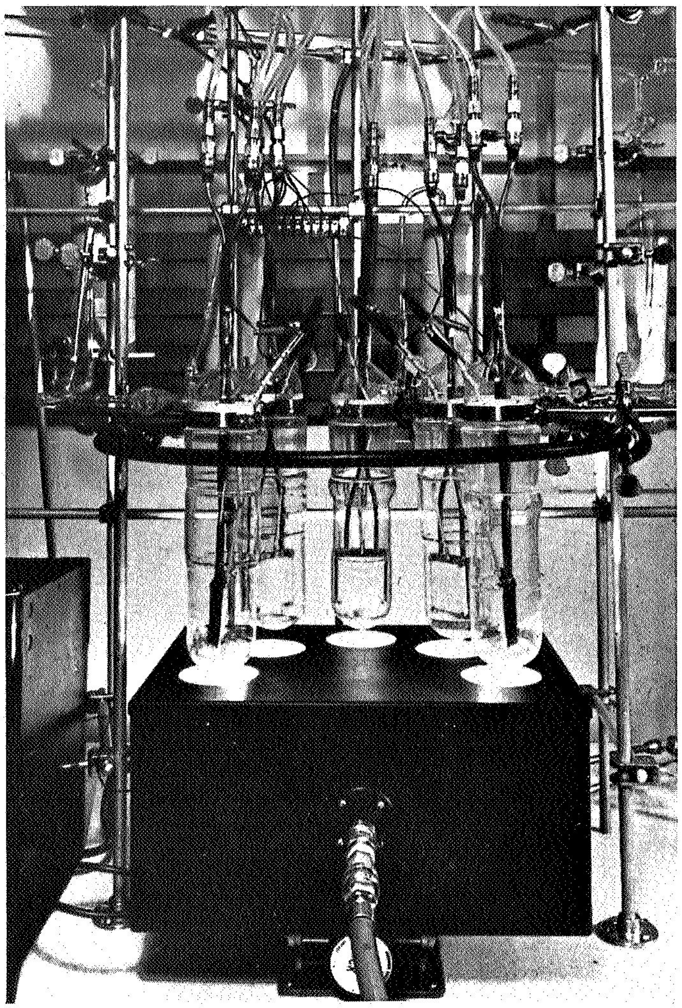


Fig. 8. Ultraviolet test apparatus

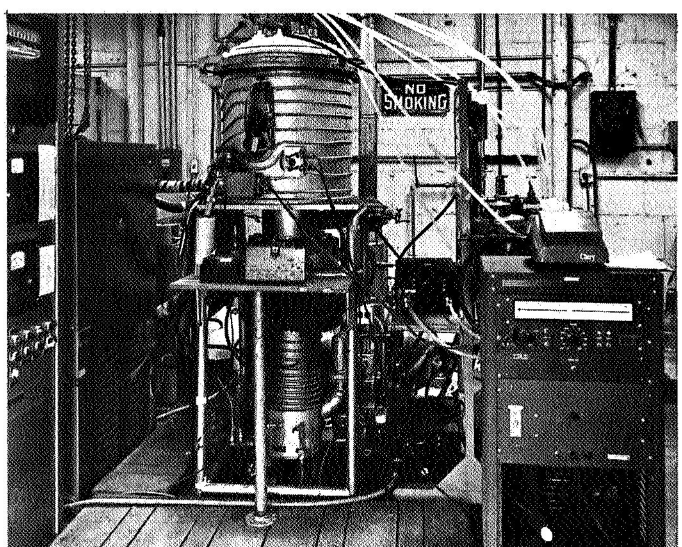


Fig. 9. Test apparatus used at Armour Research Foundation

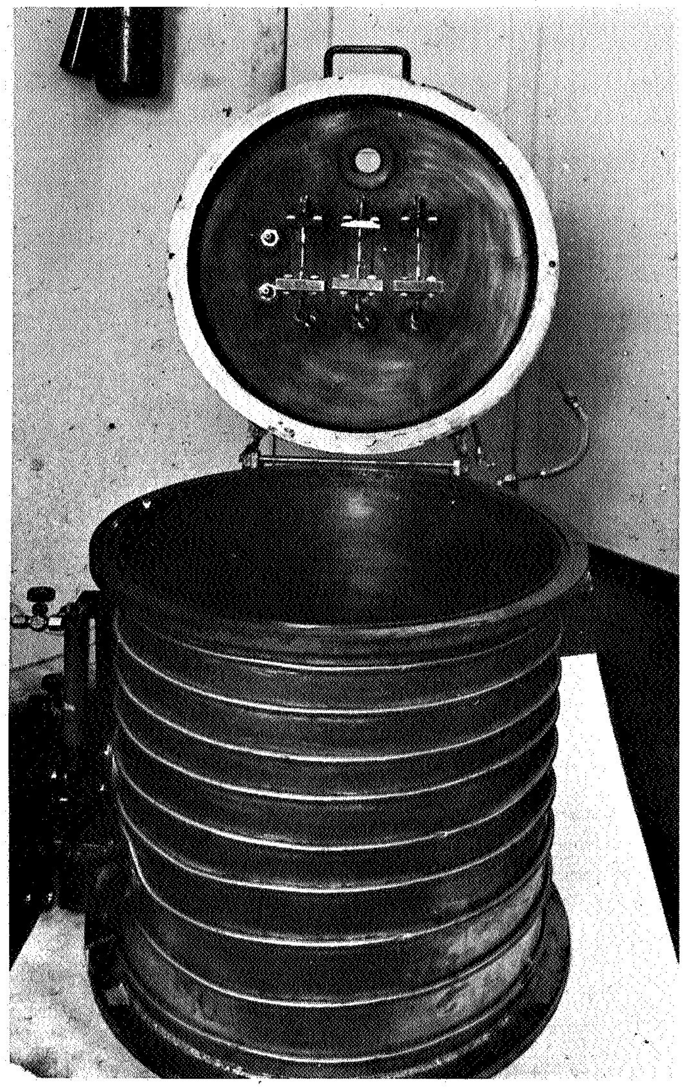


Fig. 10. Chamber of Armour test apparatus

specimens are mounted uncooled on a wheel, shown in Fig. 11. The wheel is actuated by a Geneva drive which pauses at every specimen station so that a thermistor can sense the temperature. This gives an indication of the changes occurring during the test run. Incidentally, it is possible to get away without water-cooling the specimens because the light source itself is water-cooled and the water filters out much of the radiation above 0.4 μ.

The balance of the talk will be concerned with some of the data that have been generated by Armour. The engineering requirements for the white paint are shown in Fig. 12. Figure 12a shows the principal attributes required of the white temperature-control paint for use in a space environment. Of equal importance, from an engineering point of view, are the requirements prior to

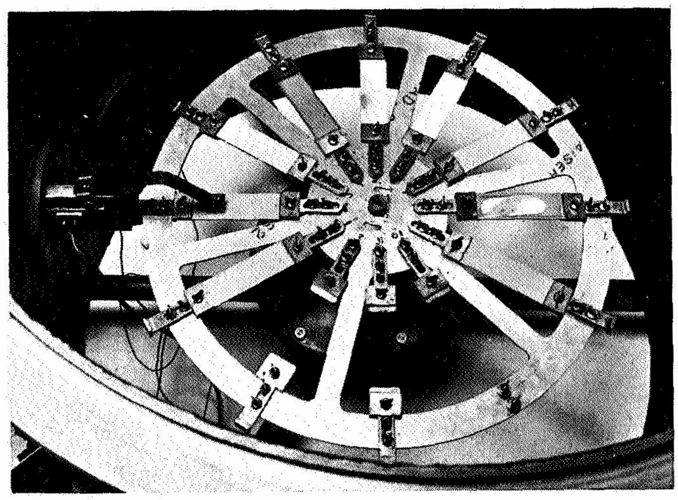


Fig. 11. Test specimens mounted on wheel in Armour apparatus

A SPACE ENVIRONMENT

1 SOLAR ABSORPTANCE (α_s) ≤ 0.35
2 EMITTANCE AT 300°K (ϵ_r) ≥ 0.80
3 STABILITY OF OPTICAL PROPERTIES UNDER CONTINUOUS SOLAR EXPOSURE
   a $\Delta\alpha_s$ ≤ 5% IN 6 MONTHS
   b $\Delta\alpha_s$ ≤ 10% IN 2 YEARS
IMPLICIT STABILITY TO OTHER SPACE ENVIRONMENTAL FACTORS
   a VACUUM - 10^{-14} mm Hg
   b TEMPERATURE - 300°F TO +140°F
   c ELECTROMAGNETIC RADIATION < 2000 A° (0.2 μ)

B GROUND ENVIRONMENT

1 APPLIED BY BRUSHING OR SPRAYING
2 AIR DRYING (NORMAL AMBIENT TEMPERATURES)
3 APPLIED AS MULTIPLE COATS (NO SOFTENING OR CRAZING OF PREVIOUS COATS)
4 COMPATIBLE WITH VARIOUS SUBSTRATES SUCH AS
   Al
   Mg
   Au
   FIBERGLASS - EPOXY COMPOSITES
   OTHER PLASTICS
5 NO CHANGE IN SIX MONTHS STORAGE IN PASADENA OR 1 MONTH AT CAPE CANAVERAL (PARTIAL SHELTER)
6 MUST WITHSTAND STERILIZATION TREATMENTS
   HEATING 24 hrs AT 275 °F (135 °C)
   10 hrs IN ETHYLENE OXIDE GAS NEAR ROOM TEMPERATURE
7 MUST STAY ON DURING THERMAL SHOCK (+200 TO -100 °F AT ≤ 50 °F/min)
8 MUST BE ABLE TO BE CLEANED TO REMOVE:
   FINGERPRINTS
   GREASE
   VACUUM PUMP OIL
   MISCELLANEOUS DIRT

Fig. 12. Principal attributes of "white" temperature-control paint

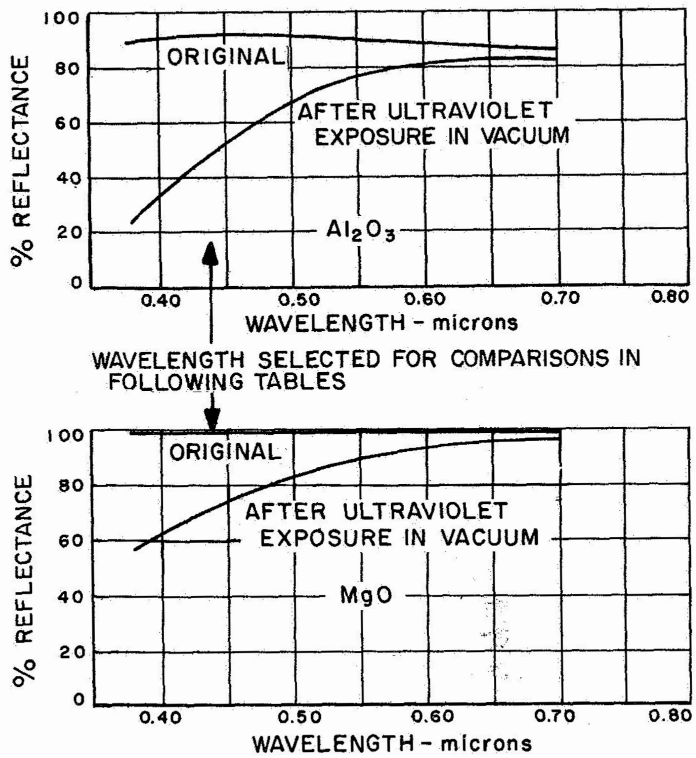


Fig. 13. Typical changes in spectral reflectance due to ultraviolet radiation in vacuum

launch, shown in Fig. 12b. When you consider that we must meet all, or most, of these requirements in a white paint for spacecraft use, the task of developing a suitable paint is a formidable one indeed. Some of the experimental results obtained to date by Armour are summarized in Fig. 13 through 16.

Figure 13 shows plots of reflectance vs wavelength before and after ultraviolet exposure for two typical pigments. The behavior illustrated is characteristic. The greatest loss in reflectance is in the shorter wavelengths. There is very little loss at 0.70 μ and at longer wavelengths. The arrow on Fig. 13 indicates the wavelength that was selected for comparison of a number of pigments in the Tables shown in Fig. 14 and 15. This wavelength corresponds closely to the peak of solar spectrum. In order to determine the solar absorptance accurately, it is necessary to integrate the area under these curves. But when you recognize where the peak is, it is readily seen that the solar absorptance would be reduced considerably after this exposure. This exposure, incidentally, was 50 hr at 1½ Suns; in other words, equivalent to 75 hr in Earth orbit. When you consider that we are looking for paints that are stable for at least four months, and in the near future, for paints that are stable for two years, you can see that these results are rather distressing.

MATERIAL[a]	REFLECTANCE AT 0.44 μ (4400Å)			NOTES
	ORIGINAL	AFTER UV EXPOSURE[b]	CHANGE	
Sb_2O_3	92.5	36.5	56	
Al_2O_3	93.5	49.5	44	
$2PbCO_3 \cdot Pb(OH)_2$	93.5	52.0	41.5	
MgO	98.5	71.0	27.5	REAGENT GRADE
$MgAl_2O_4$	97.5	70.0	27.5	(SPINEL)
ZrO_2	94.0	65.5	28.5	ULTRA-HIGH PURITY
TiO_2	–	–	SEVERE DARKENING	c

[a] HIGHEST PURITY GRADE OF MATERIAL READILY AVAILABLE WAS SELECTED FOR TESTING
[b] 50 HOURS AT 1.5 SUNS UNLESS OTHERWISE NOTED
[c] IMPOSSIBLE TO FORM SATISFACTORY COMPACT, NO REFLECTANCE MEASURE MADE
[d] 100 HOURS AT 2 SUNS

Fig. 14. Changes in reflectance of white pigments due to ultraviolet radiation in vacuum

MATERIAL[a]	REFLECTANCE AT 0.44 μ (4400Å)			NOTES
	ORIGINAL	AFTER UV EXPOSURE[b]	CHANGE	
BN	88.0	65.0	23	
$CaSiO_3$	86.0	58.0	23	SYNTHETIC $CaSiO_3$
	92.5	81.0	11.5	NATURAL WOLLASTONITE
SiO_2	88.5	77.5	11	
$ZrSiO_4$	86.0	78.0	8	(ZIRCON)
ZnS	91.0	89.0	2	REAGENT GRADE
ZnO	95.0	95.0	0	SPECTROGRAPHIC GRADE
SnO_2	77.0	77.0[d]	0	

[a] HIGHEST PURITY GRADE OF MATERIAL READILY AVAILABLE WAS SELECTED FOR TESTING
[b] 50 HOURS AT 1.5 SUNS UNLESS OTHERWISE NOTED
[c] IMPOSSIBLE TO FORM SATISFACTORY COMPACT, NO REFLECTANCE MEASURE MADE
[d] 100 HOURS AT 2 SUNS

Fig. 15. Changes in reflectance of white pigments due to ultraviolet radiation in vacuum

Figures 14 and 15 show results at 0.44 μ for a number of pigment materials that have been screened in the Armour program. It is interesting to note that such normally stable materials as aluminum oxide, magnesium oxide, zirconium oxide, and titanium oxide all degrade severely under the ultraviolet vacuum conditions of space. Fig. 15 illustrates that for the exposures shown, zinc oxide and tin oxide are quite stable, with zinc sulfide being quite interesting also. Of these three pigments we prefer zinc oxide because of the higher original reflectance. This, then, is the material that is our selection for a pigment for further paint development work. I would like to point out that in this connection we really do not understand the reason for the superiority of zinc oxide over some of the more chemically stable pigments shown.

A point that I would like to emphasize about the selection of zinc oxide pigment is that once we have found what appears to be a satisfactory pigment from the stability point of view, we still have to incorporate this into a suitable paint that will meet all the requirements shown earlier. Figure 16 shows some of the results, again at 0.44 μ, for binder materials and some of these binder materials combined with some of the pigments discussed in the previous two Figures. Of the two inorganic binders shown, which are the water-glass types, potassium silicate undergoes a significant but fairly small change. Note that the potassium silicate plus zinc oxide is considerably better than the potassium silicate alone. This is fortunate, and results from protection of the binder by the pigment material. The bottom three organic paints are all pigmented with zinc oxide. Note that the best of them, silicone RTV 11 plus zinc oxide, is still inferior to the inorganic paints. On the basis of these data alone, one would conclude that the obvious choice is the inorganic silicate paint. It certainly is a promising choice. However, we are pursuing very actively, attempts to develop better organic paints since from the standpoint of the requirements shown in the preceding Figures, the organic paints are much more likely to meet some of the Earth requirements than are the inorganics. The inorganics are characteristically brittle. They also undergo considerable shrinkage and are more difficult to apply.

MATERIAL	REFLECTANCE AT 0.44 μ (4400Å)		
	ORIGINAL	AFTER UV. EXPOSURE[a]	CHANGE
INORGANIC BINDERS[b]			
POTASSIUM SILICATE	94.0	88.5	5.5
ALUMINUM PHOSPHATE	95.5	74.5	21.0
INORGANIC PAINTS			
POTASSIUM SILICATE + $MgAl_2O_4$	93.5	76.0	17.5
POTASSIUM SILICATE + ZrO_2	86.0	79.5	6.5
ALUMINUM PHOSPHATE + ZrO_2	76.0	71.5	4.5
POTASSIUM SILICATE + ZnO	96.0	95.5	0.5
ORGANIC PAINTS			
Kel-F DISPERSION No. 800 + ZnO	83.0	58.0[c]	25
TEFLON FEP DISPERSION No. 120 + ZnO	84.0	73.0[c]	11
SILICONE-RTV-11 + ZnO	91.0	86.0[d]	5.0

[a] 50 HOURS AT 1.5 SUNS UNLESS OTHERWISE NOTED
[b] BINDERS DRIED AND PRESSED INTO COMPACTS SIMILAR TO THOSE FOR PIGMENTS
[c] 18.5 HOURS AT 4 SUNS
[d] 27 HOURS AT 4 SUNS

Fig. 16. Changes in reflectance of typical binder materials and paints due to ultraviolet radiation in vacuum

In closing, I have tried to give you a feeling for some of the materials problems involved in spacecraft. It is not possible in an hour's time to give more than a sampling of the problems involved in our current efforts at JPL. In the future we will be facing some rather severe problems; for instance, those involved with spacecraft carrying atomic reactors and the radiation damage associated with this. In the general areas of electric propulsion and thermionic converters, there are critical materials problems in connection with exposure to the very high temperatures required to obtain the high efficiencies that must be used in flight spacecraft.

The materials field is one in which progress in the past has been made by what we call "sledge hammer" techniques. Theory, in general, lags practical applications by perhaps 25 to 100 years. Progress is made by systematic experimental work, determining what works, and then rationalizing in an attempt to find a theory that will fit the experimental data. It is undoubtedly extremely important to develop our theoretical understanding. However, from an engineering point of view, we must face the fact that our understanding today is not really adequate.

OPEN DISCUSSION

Question: Are you measuring evaporation rate in your vacuum-ultraviolet testing work?

Answer: We are not measuring evaporation rates as such. However, all of our testing is done under sufficiently good vacuums that a problem with evaporation would reveal itself, should such a problem exist. I might also point out that all of our pigments and binders are selected on the basis of being relatively stable from an evaporation standpoint.

Question: Would you comment about the evaporation of compounds?

Answer: There are several mechanisms that are possible with compounds. For example, for zinc oxide you can consider evaporation of the oxide as such, or you can consider it from the standpoint of dissociation into zinc plus oxygen with the evaporation of the constituents. It turns out that the heat of formation of zinc oxide is sufficiently high that it cannot dissociate markedly, and that the vapor pressure of the oxide per se is sufficiently low. Again, these are theoretical considerations and not actual measurements. We expect to obtain some experimental data on this in the Armour contract, but we do not anticipate a serious problem.

Question: Could you comment about evaporation of alloys? So far your comments have been concerned primarily with pure metals.

Answer: The Langmuir equation works only for pure elements. The evaporation rate of an alloy, such as brass, will be much lower than one would predict for pure zinc. In this case the zinc, which is the volatile element, must be at the surface before it can evaporate. Once the first atom layer of zinc has evaporated, the next layer must diffuse to the surface. Since we are dealing with fairly low temperatures, the diffusion rates are low. I suspect that brass itself would work quite well at these low temperatures because the controlling rate is diffusion. However, we are still concerned about even small amounts of zinc where it might be deposited in other areas.

Question: Have you tried coating rapidly evaporating materials with slowly evaporating material?

Answer: This approach undoubtedly has some merit. However, we prefer to avoid the use of volatile materials where possible. In those rare cases where a volatile material must be used, I am sure the evaporation could be controlled through the use of a suitable coating. This approach might be necessary where higher temperatures are involved.

N67 11289

A Spacecraft Design Problem

JAMES L. ADAMS

Senior Development Engineer
Spacecraft Development Section

Today's lecture will discuss the problem of providing paraboloids of revolution in space. These surfaces are needed in spacecraft systems because of their unique focusing capability, which makes them extremely useful as communications antennas and as collectors of solar energy. The difficulties involved with integrating these dishes into spacecraft result from the necessity of making them live through the same environments and constraints which affect the spacecraft itself. These parabolas must be able to undergo the launch environment; they must be able to survive long lifetimes in space; they must be constrained by the same rigid minimum weight requirements; they must fit, or fold to fit, within the shroud geometry.

The process of arriving at these paraboloids is rather typical of the major portion of spacecraft design in that sufficient analytical techniques are not available to allow completely rigorous mathematical optimization. Those of you who are familiar with mechanics realize, I am sure, that a great amount of work has been done on analytical studies of shallow shells. Unfortunately, most of these studies have used assumptions of shallowness, thinness, and homogeneity that are seldom found in real life.

Most analytical work has depended upon very free boundary conditions. Needless to say, parabolas found in spacecraft have very unideal boundary conditions, as they must be constrained to the spacecraft and usually held down during the vibration phases of the launch. Little useful precedent exists for the design and fabrication of the parabolas, since previous customers for such shapes have had much different requirements. The result is that a good bit of ingenuity and experimentation is required to produce spaceworthy paraboloids.

The best way to demonstrate this design problem is to discuss a few examples. Let us first consider antennas. Antennas are not as critical as solar collectors, since they are concerned with longer wavelengths and the required tolerance for a given amount of focusing is much more lenient. However, as with the solar collectors, size is a desirable quality, since larger antennas have less edge effects and higher gain.

Figure 1 shows the first antenna that we will discuss. This is the antenna that was originally developed by the Jet Propulsion Laboratory for the *Ranger* spacecraft and that is being extended to the *Mariner* series. It is a 4-ft-diam. dish, weighing 6 lb, which is supported in the center. During launch this antenna is secured at six points around the rim from above and below. The vibration problems of the antenna have, therefore, been avoided to a good extent. The antenna is built of aluminum.

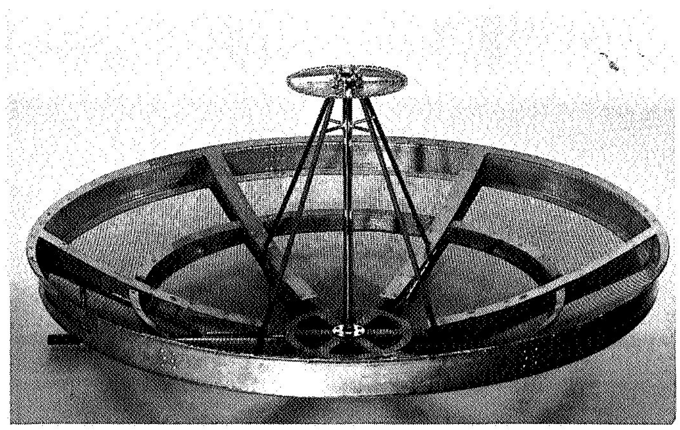


Fig. 1. *Mariner 1* high-gain antenna

The parabolic portion is ¼-in. mesh. The woven mesh is stretched over a parabolic master and then brazed. It is then clamped within the riveted aluminum structure. Problems are fairly obvious: The fabrication is not simple; the parabolic surface tends to become distorted; and the antenna is fairly heavy for the size and tolerances involved.

This antenna is not very resistant to a vibrational environment. It will be replaced on later-generation *Mariner* spacecraft by a honeycomb-sandwich dish. The parabolic

front skin will be of 0.003-in.-thick aluminum which is vacuum-formed to shape. A honeycomb layer will be formed over this skin, and the back skin applied and bonded. This antenna will also be 4 ft in diameter and approximately ½ in. thick. It will have a strengthening rim around the edge and will be filletted into a much thicker section at the center for mounting. The antenna will be secured in the spacecraft by a central mounting plus two tie-downs on the sides of the dish. This antenna will weigh 3 lb as opposed to the 6 lb of the antenna shown in Fig. 1. It will have a much higher natural frequency, the lowest significant natural frequency in this case being 80 cps as opposed to approximately 30 cps of the present one.

A 4-ft dish will fit inside of the presently existing shrouds. However, requirements have arisen for antennas larger than the size of the shroud. In this case, it is obviously necessary that they be foldable. The Jet Propulsion Laboratory did some development work on foldable antennas a few years ago. These antennas have never been flown, but they have been well tested in simulated space environments. Figure 2 shows a 20-ft-diam. antenna structure in its folded configuration. This antenna weighed 22 lb and was made from balsa wood.

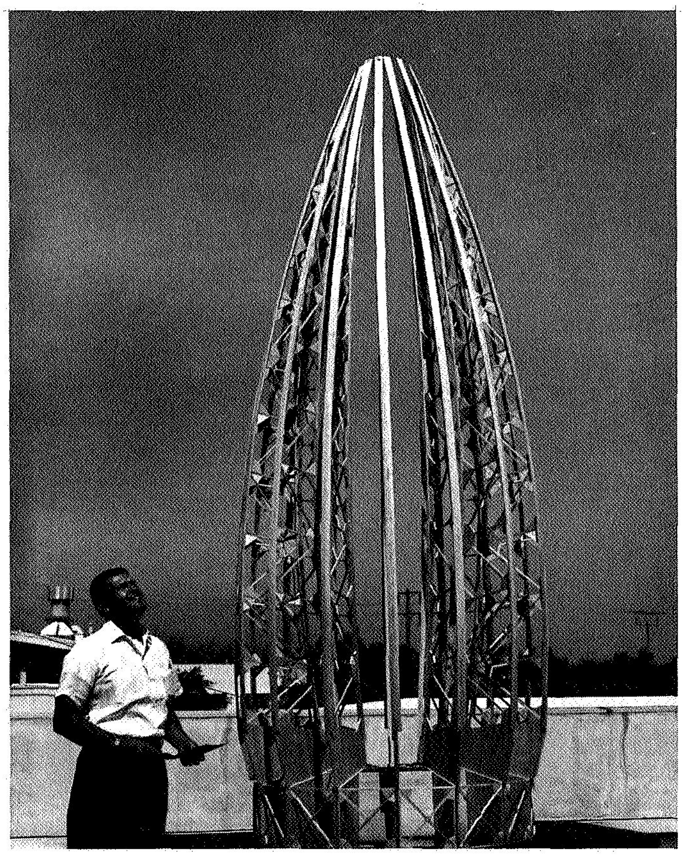


Fig. 2. Prototype 20-ft-diam. antenna, closed

Figure 3 shows the antenna structure opened. It was intended to be covered with mylar, which would be aluminum-coated to provide the electrical properties needed for the antenna. This was a very light structure weighing about 0.07 lb/ft^2.

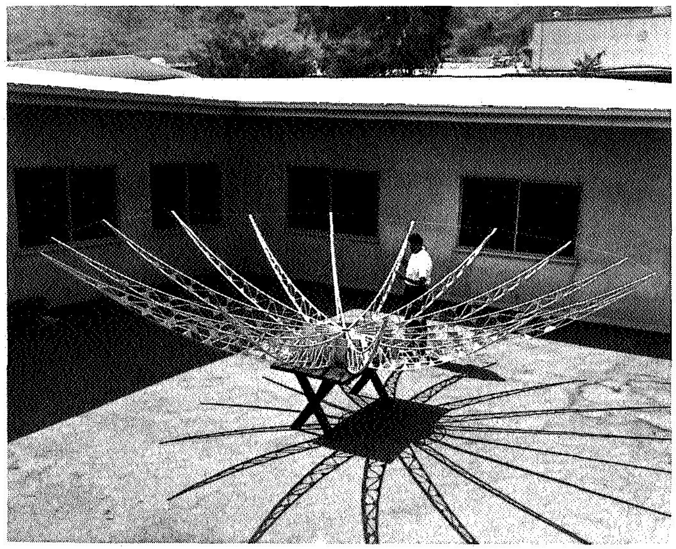


Fig. 3. Prototype 20-ft-diam. antenna, open

Figure 4 shows a 10-ft-diam. antenna, which was built in the same program, also of balsa wood. It has a layer of plain mylar in place. The antenna is shown folded up

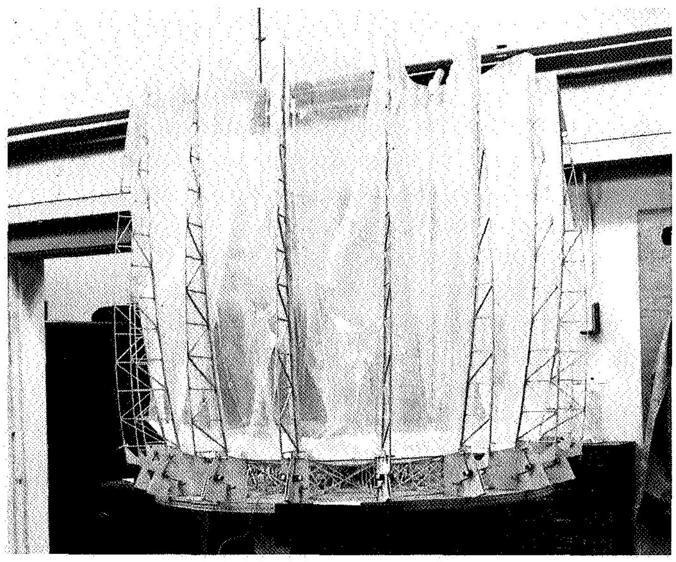


Fig. 4. Prototype 10-ft-diam. antenna, closed

on a shake table. It was successfully shaken at a level up to 10 *g* in the axial direction with no failure. It was also subjected to heat and cold, high humidity, and other flight-acceptance tests.

Figure 5 shows this antenna in an unfolded configuration. It was also shaken in this configuration up to 10 *g* with no apparent damage. The cost of this 10-ft antenna is rather interesting in that the materials cost only $7.00. There was, of course, a $700 charge for the labor involved.

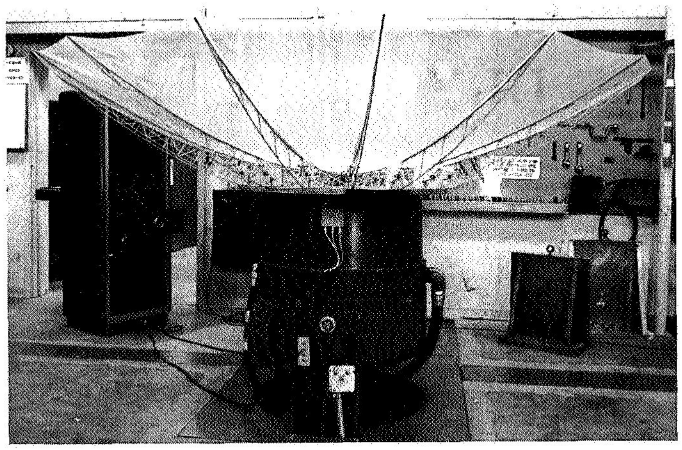


Fig. 5. Prototype 10-ft-diam. antenna, open

Let us now consider solar collectors. Solar collectors are much more of a problem, as previously stated, because of the tolerances and surfaces involved. Also, in order to obtain enough watts to fulfill all the various power functions of the spacecraft, size is a premium.

A rather typical example of a requirement for a solar collector is a solar-power system now being developed by Sundstrand Aviation, Denver, Colorado. This is a turbo-generator operated by use of solar energy. The system itself consists of a 4-stage turbine in a sealed system. The working fluid is rubidium, which is also used as a lubricant. Since this system is to be used in satellite application, heat storage must be provided so that electricity will be produced during the dark sides of the orbital passes. A heat reservoir is, therefore, provided which uses two molten salts, lithium hydride and sodium fluoride. A rotating condensor collects the condensed fluid by centrifugal scavenging. This system can be made in a variety of sizes. The original requirement was for a 15-kw device. A device of this size would require a 45-ft-diam. mirror. This is obviously a much larger mirror than any presently existing shroud could accommodate. The mirror must therefore fold.

Sundstrand Aviation is subcontracting the development of the mirror. Goodyear Aircraft Corp.[1] has proposed a foam-rigidized parabolic reflector. In order to form this reflector, a parabola is first made of aluminized mylar. Figure 6 shows this process.[2] A mylar bag is then attached to the front of the reflector, creating a large balloon, and a mylar membrane is attached to the back of the mylar reflector surface. The balloon can then be inflated in space, with the result that the reflective surface becomes parabolic. Foam is then injected between the mirror and the membrane. The nonreflective portions of the balloon are then burned away by use of a primer cord which wraps around the outside of the balloon. The foam-rigidized reflector system for the 45-ft-diam. mirror needed for the full 15-kw capability weighs 390 lb. It packages to a disk approximately 8 in. long and 8 ft in diameter. The inflated, rigidized mirror weighs approximately ¼ lb/ft². The original specifications for the mirror called for 98% of the surface to have a tangential error within ±¾ deg.

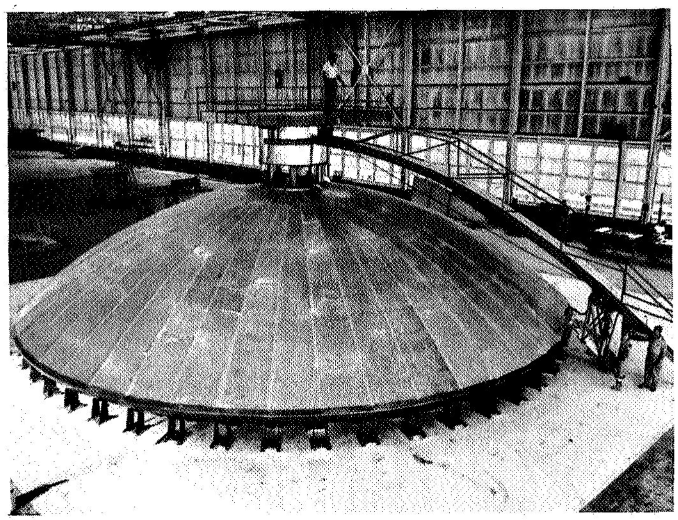


Fig. 6. Mylar reflector fabrication

Figure 7 shows the back of a 10-ft-diam. demonstration collector. There are some problems with this method. The pressure in the balloon is rather critical. Foaming is relatively unpredictable, since it is not known exactly how much foam will result from a given quantity of original material. The advantages are that the system is fairly lightweight, and the optical qualities of the front surface are very excellent.

[1]Space Systems Division, Akron, Ohio.
[2]Photographs in Fig. 6 and 7 courtesy of Goodyear Aircraft Corp.

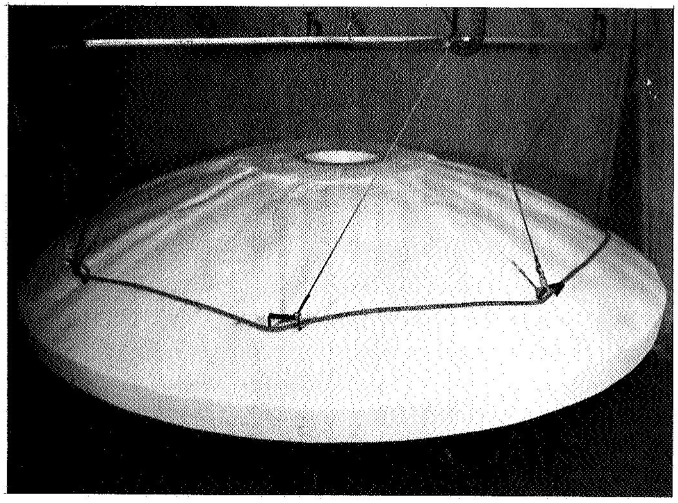


Fig. 7. 10-ft-diam. foam-filled mirror, rear view

A completely different approach to the same problem was taken by Ryan Aeronautical Co., San Diego, California. They developed a mirror which was made entirely of folding metal structure. Figure 8 shows this item in its folded configuration.[3] It is bulkier than the foam-filled one in the folded state. It requires a package that is approximately the diameter of the mirror in length, and almost a quarter of the diameter of the mirror in diameter. Advantages of this all-metal construction are that it can be checked out. It is not a one-shot device as the Goodyear balloon is. Accuracy can be tested, and then the article can be folded and unfolded repeatedly. This 10-ft prototype development weighs approximately 28 lb and consists of 36 petals all hinged at the bottom ring, pivoted at the top to a circular lazy-tong structure. The mirror is actuated by extending this lazy-tong structure by means of cables. Figure 9 shows the mirror in a partly open configuration. Figure 10 shows the mirror fully open. When the mirror is fully opened all petals are bottomed against stops, which guarantees alignment and provides a rigid structure. As can be seen from Fig. 10,

[3]Photographs in Fig. 8–11 courtesy of Ryan Aeronautical Co.

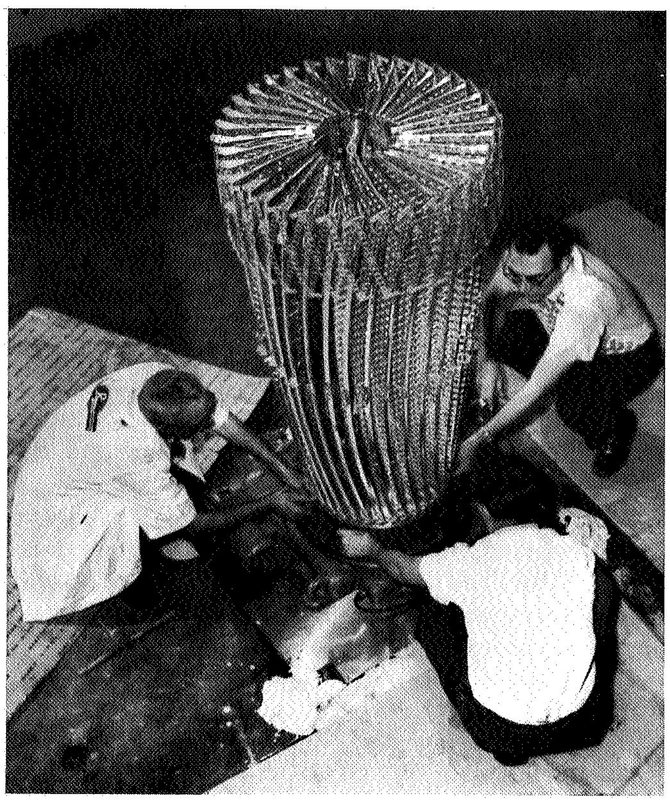


Fig. 8. Prototype metal mirror structure, folded

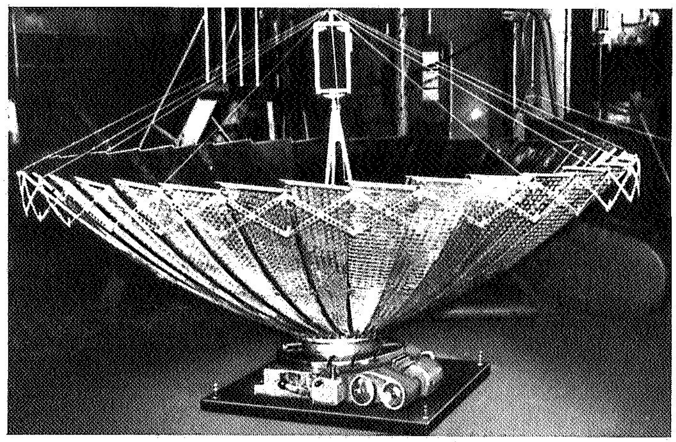


Fig. 9. Prototype metal mirror structure, partially open

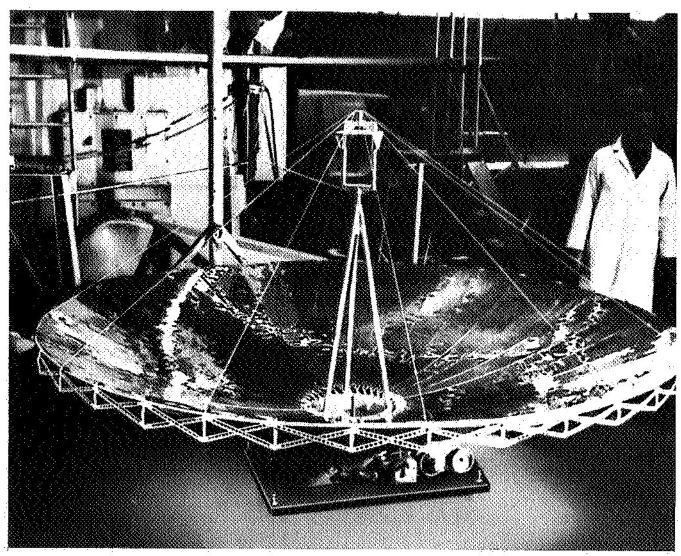


Fig. 10. Prototype metal mirror structure, fully open

the defects of the mirror as originally conceived were in the optical qualities of the surface. Ryan wanted to create a completely resistance-welded structure. They were unwilling to gamble on bonding agents because of the yearlong lifetime which was required.

The backing structure is a lattice truss built of channel strips made of 0.004-in.-thick aluminum. The original reflecting skins were only 0.005-in. thick. The result was that the backing structure welds caused distortion in the front skin. Later developments have shown that by going to 0.020-in. skin the quality of the reflecting surface can be greatly improved. Some of the redundant structure can be chem-milled away to provide the original lightness. Figure 11 shows the back of the Ryan mirror. The lattice truss members can be plainly seen.

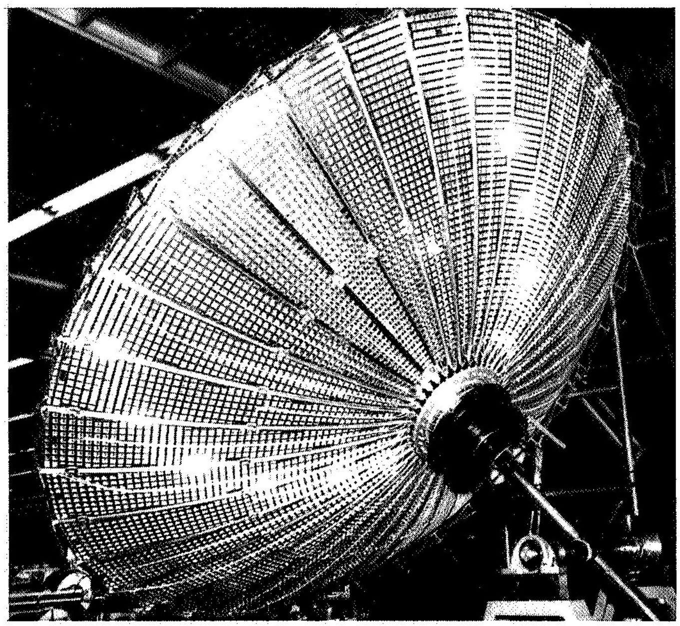


Fig. 11. Prototype metal mirror structure, rear view

A completely different approach to mirror construction has been taken by Electro-Optical Systems, Inc., Pasadena, California. They have a contract from Jet Propulsion Laboratory to provide a solar collector for a thermionic device. Electro-Optical Systems was extremely interested in optical qualities because of the nature of the energy conversion mechanism. Thermionic diodes are small in physical size and operate on a temperature gradient. It is therefore very important that their solar collectors focus the solar energy well enough so that the gradient can exist. Electro-Optical did quite a bit of experimenting with various means of fabricating lightweight mirrors. They found, generally, that forming methods such as stretch forming, explosive forming, and vacuum forming were not too successful for extremely lightweight skins because of the fact that the skins had to be put in very close physical contact with the master molds. This resulted in surface imperfections. It is very difficult to remove these imperfections by polishing on skins which are only a few thousandths of an inch thick. Electro-Optical Systems met similarly bad luck in their attempts to use backing materials with thin reflective skins. The backing materials would show through. Even such close-knit materials as foams would cause a slight orange-peeling of the mirror surface. They therefore settled upon electroforming as a fabricating method. In electroforming, the part is built up by plating onto a mold, which is subsequently removed.

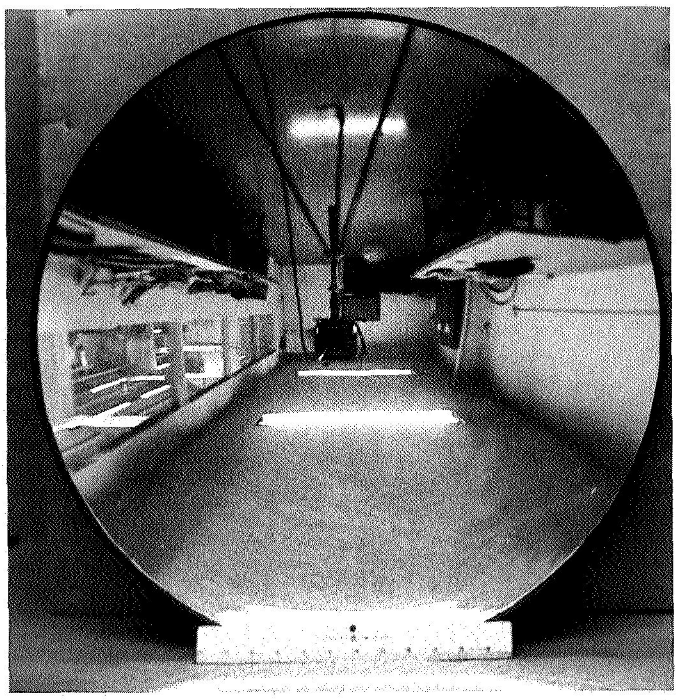
There are several difficulties involved with electroforming, the major one probably being the separation of the finished work from the original master. However, by research on the eccentricities of the plating process, Electro-Optical was able to solve the problems to a great extent. One major difficulty was controlling the internal stresses in the plating. If a part is to be successfully removed from the master, these stresses must be neutralized. The master, of course, must be extremely smooth, contamination must be removed, and temperatures and currents controlled very closely.

Figure 12 shows a 24-in. mirror manufactured by Electro-Optical Systems.[4] This mirror was made by plating a plastic master. After the plating was finished, strengthening was required in order to give the mirror sufficient structural strength. A very efficient way of strengthening a mirror of this type is by reinforcing the edge. Therefore, a torus was electroformed and cemented onto the mirror. This was not too successful a development, because the torus tended to warp the mirror.

Different methods of forming tori were experimented with. An early attempt was to weave wire over an aluminum torus, place the woven-wire torus on the master, and then plate over the woven wire. The woven-wire torus would then be encapsulated into the back of the master. After the plating was completed, the aluminum torus would be dissolved from inside of the woven wire. This made a very light, very rigid structure. Unfortunately, the stress concentrations were bad where the wire met the mirror. Show-through of the wire torus was also very bad.

Electro-Optical has had their greatest success with essentially one-piece-type construction, such as is shown

[4]Photographs in Fig. 12–16 courtesy of Electro-Optical Systems, Inc.

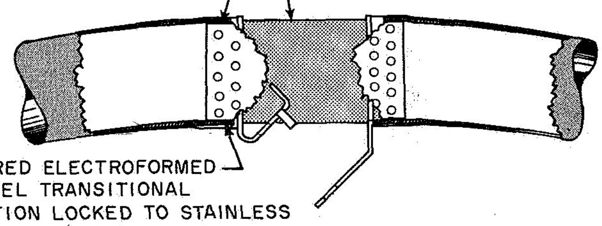


a. Front view

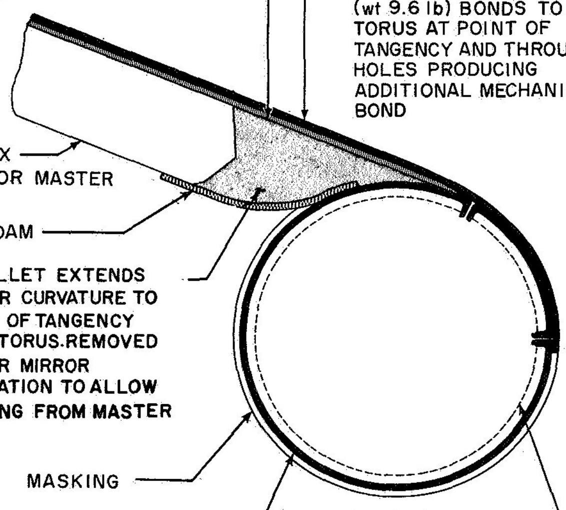
b. Rear view

Fig. 12. 24-in. electroformed mirror

in Fig. 13. In this fabrication method, a torus is formed by plating an aluminum-tubing torus, and then dissolving the aluminum away from the inside with acid. Nickel is used as the plating agent and these tori are generally plated to a thickness of approximately 0.012 in. After the torus is completed, the master is plated to the thickness which will provide a good front surface (typically 0.004 or 0.005 in.). The torus is then placed on the back of the mirror and the plating is continued. The plating will encapsulate the torus and result in an essentially one-piece structure.



a. DETAIL OF BRACKET AND TRANSITION



b. MIRROR–TORUS STRUCTURAL DETAIL

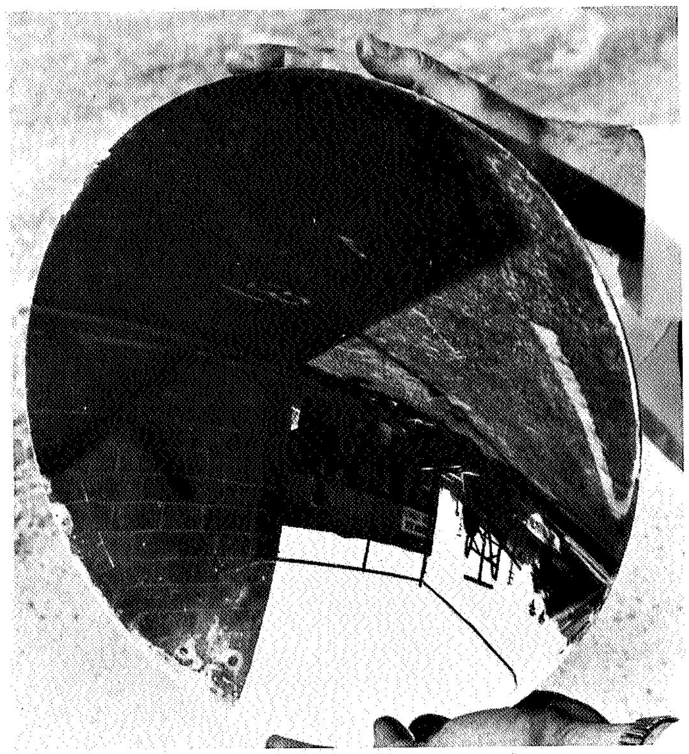
Fig. 13. Mirror construction details

Figure 14 shows a 5-ft mirror that was produced in this manner. The reflective face is deposited aluminum. The mirror weighs slightly over 14 lb; the torus itself weighs 5 lb and is 0.012 in. thick; the mirror skin is 0.008 in. thick.

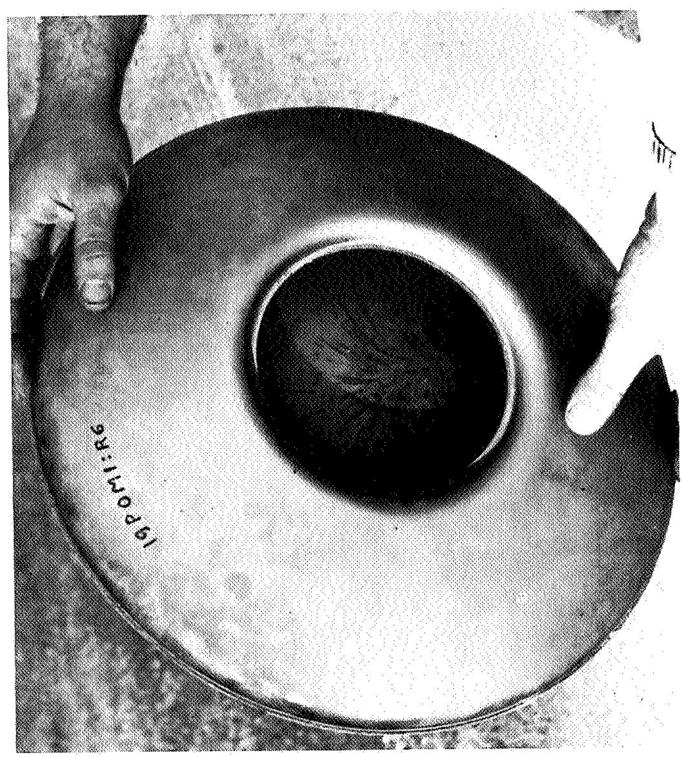
Fig. 14. Complete 5-ft-diam. electroformed concentrator with integral mounting brackets

The final development, of course, includes mounting brackets in the torus, which add an additional 2 lb. The final mirrors are also thickened slightly around the edges, tapering from 0.008 in. in the center to 0.011 in. at the edge. In tests at the Jet Propulsion Laboratory, a 5-ft mirror has been shaken successfully in the axial direction. Unsymmetrical and transverse tests have not yet been completed, but these are expected to be less crucial than the axial test.

Electro-Optical Systems has also experimented with other types of mirrors than the toroidally supported one. Figure 15 shows a small monocoque mirror. This type of mirror has the advantage of being supportable at the center. The stresses are still distributed at the edge of the mirror. This mirror was made over a parabolic master, as was the larger mirror. After the front skin was plated, a form was made which would fit over the plating. This form was sprayed with cerro metal, which is an alloy that melts at the temperature of boiling water. The sprayed cerro metal was then separated from its form and placed



a. Front view



b. Rear view

Fig. 15. Electroformed monocoque mirror

on the back of the plated mirror. Nickel was then plated over the cerro metal form and over the exposed edge of the mirror. Hot water was then flushed over the completed product and the cerro metal melted out. The result was a strong one-piece monocoque mirror. Mirrors of this sort have been made as thin as 0.002 in. and found to be structurally quite strong.

Figure 16 shows a one-piece monocoque electroformed petal for a folding mirror such as the Ryan Aeronautical mirror. This petal is a 20-deg segment 19 in. long, with a 0.003-in. backing skin. Thickness varies from 0.002 to 0.006 in. on the front face. The specific weights generally are around 0.2 lb/ft^2 of mirror surface.

The examples that have been presented are a few approaches that have been taken to solve the problem of providing parabolic surfaces in space. As I have tried to show, this is an area where imagination and creative approaches must be relied upon. The design and fabrication of parabolic surfaces for spacecraft use is an example of the ever-familiar situation where the need has out-raced the technical know-how. As such, it is a typical spacecraft design problem.

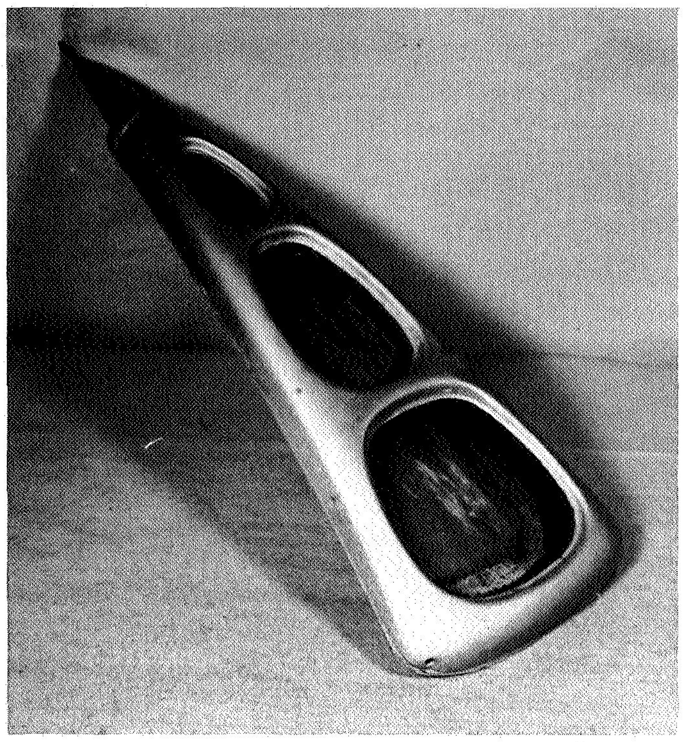


Fig. 16. Electroformed mirror petal